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Aircraft Control—
A Survey.

By W. TREVARTON TRUSCOTT, B.Sc. (Eng.).

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CONTENTS.

AIRCRAFT CONTROL — A SURVEY.

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I. INTRODUCTION.

It is intended to discuss the problems of control in general and to review the methods adopted to overcome them.

No attempt is made to develop elaborate theoretical analyses or to reproduce type details found in maintenance manuals and manufacturers' handbooks ; but rather an effort to deal with the subject simply, yet comprehensively in scope and fundamentally in manner. The use of mathematics has been purposely discarded except for a few cases which have been incorporated mainly for the benefit of students. These are illustrated by practical numerical examples.

II. THE NEED AND THE PROBLEMS.

In flight an aeroplane has six degrees of freedom. That is, it can have six independent components of velocity ; linear velocities *along* the longitudinal, lateral and normal axes ; and angular velocities *about* the longitudinal, lateral and normal axes termed rolling, pitching and yawing, respectively. The axes, the origin of axes—the aircraft centre of gravity—and component velocities are shown in Fig. 1.

Four basic flying controls are usually needed for successful flight manipulation. These are :—

1. A *forward control* effected by variation in magnitude of the propulsive or drag force.
2. A *rolling motion control.*
3. A *pitching motion control.*
4. A *yawing motion control.*

} Generally effected by changing the relative positions of parts of the aircraft surface by means of movable flap arrangements (*i.e.*, control surfaces).

An aeroplane is said to have satisfactory control characteristics if the pilot can perform the functional manoeuvres without incurring undue strain.

A number of factors often make this seemingly straightforward specification somewhat difficult to fulfil.

The common factor for all types of flying control systems is the capacity of the pilot : the capacity of the pilot, amongst other

things, in terms of his muscular strength; his reaction time to visual, aural and tactual stimuli; and his mental and physical fatigue characteristics.

There are, too, a number of laws which form the basis of all flying control design.

The control hinge moment, and hence the stick force in a simple control system, is proportional to the square of the equivalent air speed. (Equivalent air speed = V_e = E.A.S. = true air speed × square root of relative density of air). Thus if a particular stick force is 50 lbs. at 100 m.p.h. E.A.S. the corresponding stick force at 200 m.p.h. E.A.S. would be 200 lbs. This tends to excessive stick forces at high speeds with little resistance to stick movement at low speeds; the limits being inadequate muscular strength of the pilot at one end of the scale and ultra-light controls with lack of aircraft response to control movement at the lower end of the speed scale.

The control hinge moment also varies according to the cube of a linear dimension of the aircraft. Again let a particular stick force be 50 lbs. under certain conditions. For a similar aircraft but of twice the size, then, for the same conditions and airspeed the corresponding stick force would be 400 lbs. Thus similar control system design would not be suitable for both large and small aircraft.

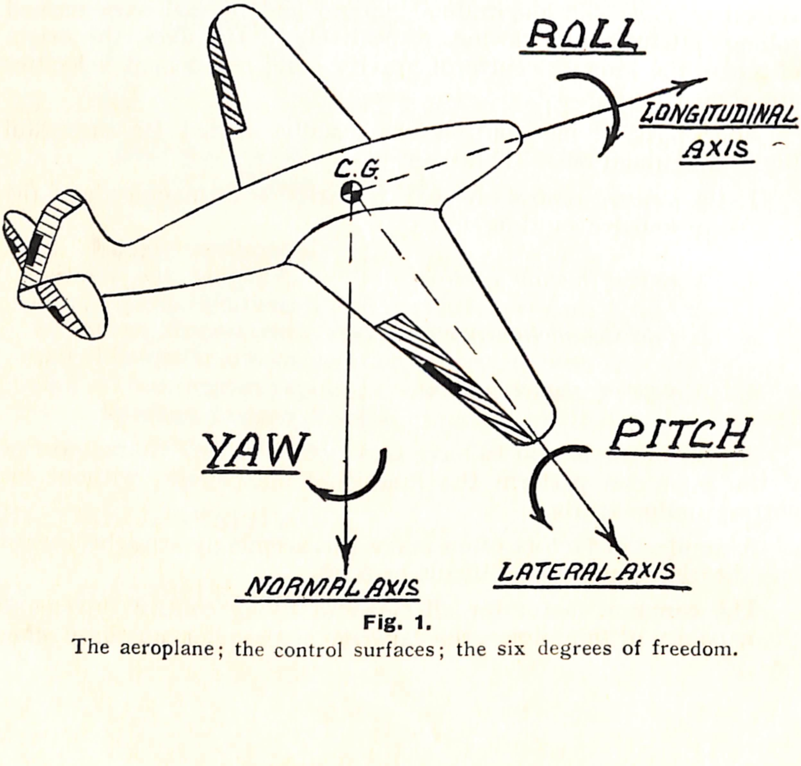


Fig. 1.
The aeroplane; the control surfaces; the six degrees of freedom.

The designer, too, may have to take into account the effects of wide ranges of temperature variation in, say, steel control runs fitted in aluminium alloy structures. This would involve considerations of differential expansions and contractions due to the diverse coefficients of expansion of aluminium alloy and steel.

With aircraft designed to fly at high altitudes he may also be beset with the added complications of control run installation through pressurised cabins.

Aeroelastic and compressibility effects are also dominating factors in influencing control surface and system design.

In the face of these many difficulties and also in view of the fact that there has not been complete unanimity in the many conclusions emanating from test and research reports and mathematical analyses, is it surprising that control surface design methods are inclined to be somewhat intuitive rather than strictly scientific in manner?

III. THE HUMAN PILOT.

The efficient functioning of any control system, however well designed and maintained, depends primarily on the operator, the pilot. It is thus appropriate to discuss the capacity and characteristics of the human pilot in some detail.

Sense of Balance.

In flight pilot and aircraft constitute an entity in space, a system of reference, relative to which the earth moves.

Under visual flight conditions the relative position and movement of earth and aircraft are discerned by the pilot through the harmonious integration of the visual, vestibular and deep muscular senses which together make up his sense of balance. The vestibular and deep muscular senses are an aid to balance when linked with a strong visual stimulus but under adverse visual conditions they are unreliable, giving way to sensory illusions.

(*a*) The Visual Sense.

The visual sense, provided by the eye, is the most important of these sense organs. It identifies position relative to other bodies and to a datum line, the horizon.

(*b*) The Vestibular Sense.

The vestibular organ which provides this sense consists essentially of three liquid filled tubes and a common pouch whose inner

walls are sensitive to the relative movement of fluid. The organ is situated in the inner ear.

1. One tube lies in the lateral vertical plane and records accelerations in roll.
2. The second tube lies in the vertical fore and aft plane and records accelerations in pitch.
3. The other tube lies in the horizontal plane and records accelerations in yaw ; whilst
4. The pendulous effect of the fluid level in the lower part of the pouch records the direction of apparent gravity.

When vision is restricted the sensory transmissions to the brain from the vestibular organs are unreliable. Resulting sensory illusions include those caused by the following :—

A. With sustained rotation of the aircraft in a particular plane at constant angular velocity there is no relative movement of fluid within the tube in that plane, so no sense of impression of angular motion is given.

B. On rapid cessation of angular motion the momentum of the fluid in the tube in the plane of rotation will cause it to surge onward giving rise to an impression of rotation in the opposite sense.

C. In a correctly banked turn the line of action of apparent gravity stays normal to the wings and the liquid level in the pouch thus remains parallel to the wings and no feeling of bank is transmitted to the pilot's brain.

D. In a flat turn, however, the application of a lateral acceleration force causes an inclination of the fluid level in the pouch giving the pilot an illusory impression of banking.

(*c*) The Deep Muscular Sense.

This is a sense whereby the effect of the loading on inner parts of the human body gives an indication of its attitude. The combination of aircraft and gravitational accelerations may give rise to illusory impressions being transmitted to the brain when this sense is not co-ordinated with, and dominated by, the sense of vision.

These limiting factors indicate that adequate aircraft control by a human pilot cannot be achieved under conditions of limited visibility by recourse simply to his natural senses, due to the unreliable and illusory nature of their responses.

The need, then, is for artificial means of presenting to him the attitude and rates of motion of the aircraft in and about the axes and planes of reference. This is provided in the form of dial instruments.

Acceleration or 'g' Sensitivity.

In aeronautics accelerations are referred to in terms of the gravitational constant of acceleration 'g'. An acceleration of '2g' of a body would induce a force on the body of twice the magnitude of that due to gravity.

Centripetal accelerations normal to the flight path, which occur when the line of flight is curved, are of greater magnitude than those experienced along the line of flight except, possibly, under adverse emergency landing conditions when up to '20g', or so, might arise.

Under visual flight conditions small accelerations assist the pilot in maintaining control. In manoeuvres involving larger 'g' the inertia forces caused by these accelerations may produce deleterious effects on the pilot in the form of blackouts and red-outs although under these circumstances he tends to become a safety valve, or human 'g' restrictor, in that knowing his own 'g' limit he does not tolerate loading of higher 'g' severity on the aircraft structure.

A 'black-out' is a temporary loss of sight, and possibly also of consciousness, caused by the draining of blood from the head following the imposition of inertia forces in the direction head to foot. The black-out threshold varies according to the time of application and severity of the acceleration. For example, a pilot may withstand '8g' for a few seconds but black-out under '4g' sustained for several seconds.

A 'red-out' brings stars before the eyes with possible unconsciousness and is caused by the rush of blood to the head following the imposition of inertia forces in the direction foot to head under what is termed 'negative g' conditions. The red-out threshold is somewhat lower than that for black-out and the after effects are much more pronounced.

As the inertia forces are more severe along the normal to the flight path arrangement of the pilot's head to foot line, to be at right angles to this, that is having the pilot in the prone position, raises the threshold. Raising of the threshold too may be accomplished by the wearing of 'anti-g' suits, suits which apply a pressure progressively increasing with 'g' severity to the lower parts of the body.

Time of Response.

The time lag between the deviation of an aircraft from a specified setting and its return depends on :—

1. The time taken for the pilot to perceive the deviation.
2. The time taken for the pilot to physically react after perception.

3. The time taken for the aircraft to respond to control application.

If the motion to be corrected is oscillatory and of short period compared with the total reaction time the effect of purely reactive control 'correction' may be to increase the deviation and the motion may only be smoothed out by skilful anticipatory control movement, or, if the frequency of oscillation is sufficiently great it will be uncontrollable by the pilot.

Fatigue.

The prevalency of well established but, as has been shown, possibly illusory instincts, coupled with the lessened power of response to stimuli brought on by pilot fatigue, may greatly affect the efficiency of control of an aircraft.

Features of the flying control and other systems of an aircraft which reduce the possibility of pilot fatigue include :—

1. The elimination of excessive flying control forces.
2. Harmony of flying control forces.
3. Provision of an adequate trimming system.
4. Avoidance of over-sensitive controls.
5. Good cockpit control and instrument layout.
6. An adequate oxygen supply at altitude.
7. The fitting of an automatic stabiliser.
8. The inclusion of an auto-pilot.

IV. THE SIMPLE FLAP.

The flap is a movable part of an aerofoil which, in its neutral position, constitutes the rear or forward portion of the normal aerofoil profile. A simple trailing edge flap is illustrated in Fig. 2a.

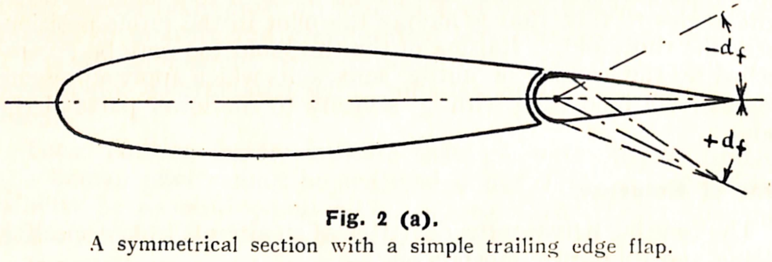


Fig. 2 (a).
A symmetrical section with a simple trailing edge flap.

Lift.

Deflection of the flap not only changes the angle of attack of the flap itself but also alters the camber of the section as a whole.

Thus the lift increment due to flap deflection is more than would be obtained by turning the flap alone in the absence of the main section, and is not built up entirely on the flap but is distributed along the chord.

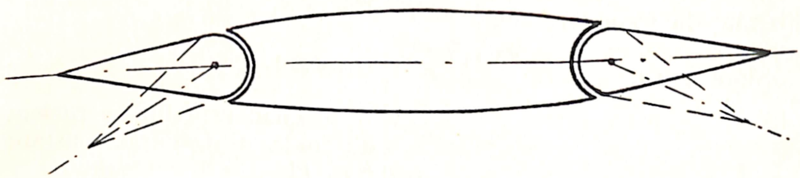


Fig. 2 (b).
A "high speed" section with trailing edge and leading edge flaps.

In the case of the section with leading and trailing edge flaps (Fig. 2b) the effect is to introduce considerably more camber to improve the lift properties of sections such as those of sharp nosed thin symmetrical families whose basic poor lift characteristics are well-known. The crescent-winged Handley-Page Victor incorporates leading edge flaps—droop snoots—fundamentally similar to that shown in Fig. 2b.

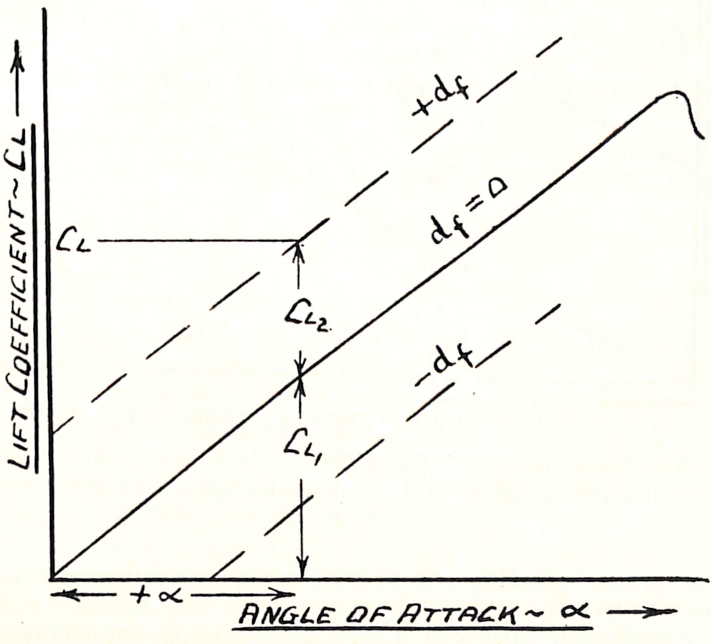


Fig. 3 (a).

For a symmetrical aerofoil with a simple trailing edge flap at zero incidence the relationship between the lift coefficient and the angle of attack is as shown in Fig. 3a. From zero angle of attack to near the stalling angle a practically linear relationship exists between the coefficient of lift and incidence. Referring to Fig. 3a, this may be expressed mathematically as :—

$$\text{Slope of Curve} = \frac{Cl_1}{\alpha} = \text{a constant.}$$

$= $ rate of change of lift coefficient with angle of attack at constant flap angle.

$= a_1$ (say).

$$\therefore \quad Cl_1 = \alpha \times a_1 \qquad (1)$$

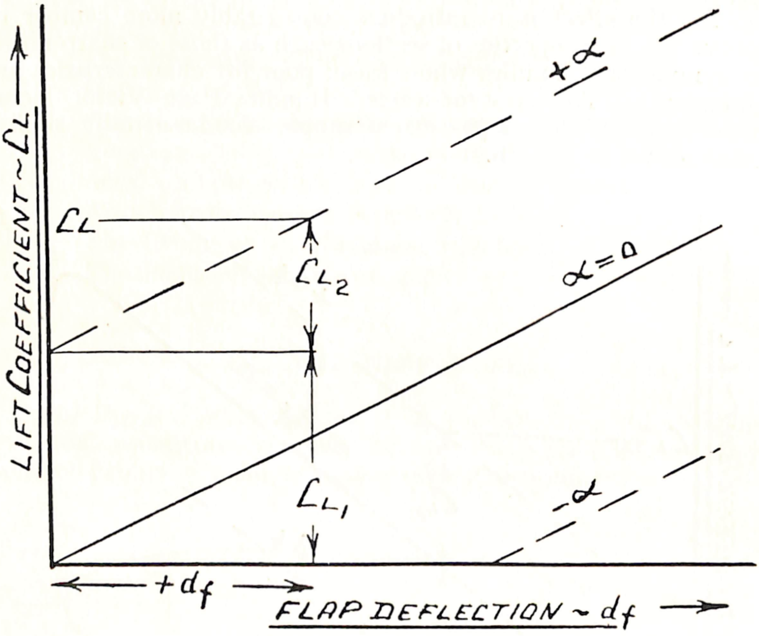


Fig. 3 (b).

The lift characteristics of a main section having a simple trailing edge flap.

Curves corresponding with positive and negative flap deflections are parallel to but displaced above and below the original line.

For particular angles of attack with varying flap deflection, Fig. 3b results. The graph shows that the coefficient of lift also

varies linearly with flap deflection for a constant angle of attack. Reference to the 'plus alpha' line of Fig. 3b gives that :—

$$\text{The Slope} = \frac{Cl_2}{d_f} = \text{a constant.}$$

$$= \text{rate of change of lift coefficient with flap deflection at constant angle of attack.}$$

$$= a_2 \text{ (say).}$$

$$\therefore \; Cl_2 = d_f \times a_2. \qquad (2)$$

Observation of Figs. 3*a* and 3*b* show that the total lift coefficient brought about by a flap deflection (d_f) coupled with a main surface angle of attack (α) is :—

$Cl = Cl_1 + Cl_2$, which yields from (1) and (2) that

$$Cl = \alpha a_1 + d_f a_2 \qquad (3)$$

Very approximate values of the lift slope (a_1), corresponding to low subsonic, high subsonic and supersonic flight are given in the graph in Fig. 4. For aerofoils used as tail surfaces the 'end plate effect' of a fin (or fins) on a horizontal tail surface is to increase a_1; but the effects of fuselage blanketing of the horizontal tail surface and of rudder travel cut-outs in the elevator is to decrease the rate of change of lift coefficient with angle of attack.

The ratio of a_2 to a_1 is independent of aspect ratio and taper ratio. It depends on the ratio of flap chord to overall chord, to the hinge gap leakage and to the trailing edge angle.

Equation 3 shows that the same increment of lift produced by a flap deflection (d_f) could be achieved by a change in angle of attack of :—

$$\alpha = \frac{a_2}{a_1} \times d_f \qquad (4)$$

Taking a typical value of $\dfrac{a_2}{a_1} = 0{\cdot}5$, then a flap deflection of, say, 12° would produce the same lift increment as a change in main surface angle of 6°.

Equation 4 and this example show that an all-moving and adjustable section (*e.g.*, horizontal tail surface) would have exceedingly powerful control characteristics.

The effect of simple flaps on the lift properties of an aerofoil has been dealt with in some detail for it forms the basis of control surface mechanics. The elevator, rudder and aileron are simply flaps, modified to suit their particular needs, which induce aerodynamic forces at distances from the centre of gravity of the

aircraft. These generate moments—moments applied by the pilot—controlling the movement of the aircraft about the various axes.

Drag.

The effect of flap deflection is to increase the profile drag of the aerofoil always, whereas the effect on the induced drag is to increase it if the flap deflection has caused an increment in lift but to decrease it if the deflection has caused a decrement in lift.

The most practical outcome of this will be considered in the section on ailerons.

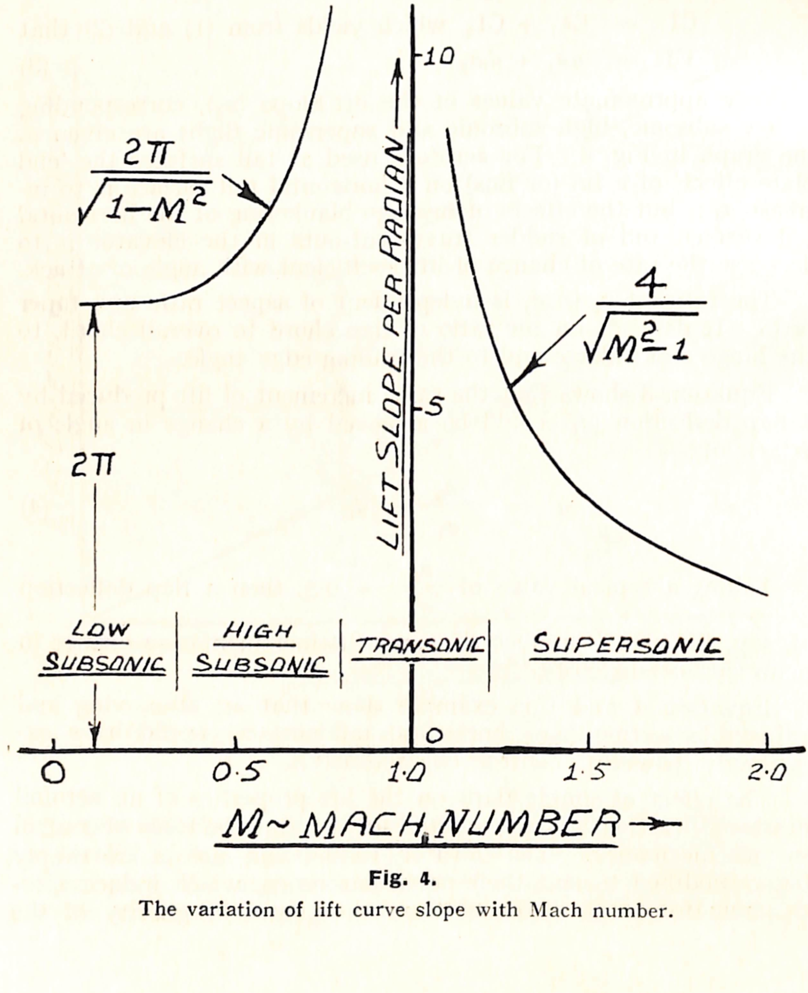


Fig. 4.

The variation of lift curve slope with Mach number.

Hinge Moment (Fig. 5a).

The flap hinge moment, H_f, is the moment necessary to deflect, or keep the flap deflected. It is the moment of the resultant air pressure on the flap about the hinge axis. The magnitude of the flap hinge moment is given by :—

$$H_f = \tfrac{1}{2} \rho_o V_e^2 S_f c_f Ch_f \text{ lbs. ft.} \qquad (5)$$

where V_e = equivalent air speed in ft./sec.
S_f = flap plan area aft of the hinge line in sq. ft.
c_f = flap chord aft of the hinge line in ft.
Ch_f = flap hinge moment coefficient.
ρ_o = density of air at sea level (0·002378 slugs/cub. ft.).

As with the lift coefficient it can similarly be shown that, over a limited range, the flap hinge moment coefficient varies linearly with angle of attack and also with flap deflection. It may thus be expressed mathematically as :—

$$Ch_f = b_1 \alpha + b_2 d_f \qquad (6)$$

where b_1 and b_2 are constants.

If the aerofoil is not symmetrical (*i.e.*, it possesses camber) then another constant (b_o) is added to the right-hand side of equation 6.

Equations 5 and 6 yield :—

$$H_f = \tfrac{1}{2}\rho_o V_e^2 (S_f c_f) (b_o + \alpha\, b_1 + d_f b_2) \text{ lbs. ft.} \qquad (7)$$

Equation 7 shows that for a particular angle of attack and flap deflection the flap hinge moment is proportional to the square of the indicated air speed and also to the cube of a linear dimension ($S_f c_f$ term) of the aircraft. This equation, then, justifies two of the laws mentioned in Section II.

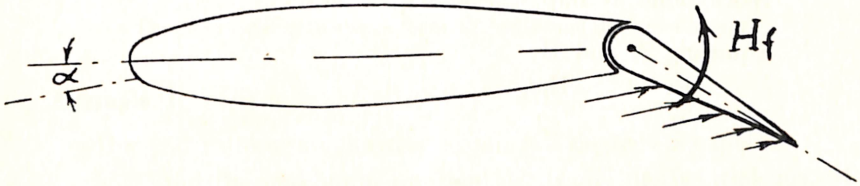


Fig. 5 (a).
The hinge moment.

The Tab.

The tab is an auxiliary adjustable flap inset in the trailing edge of a movable control surface.

For a section with a movable flap and tab arrangement, as shown in Fig. 5*b*, equation 7 extends to :—

$$Ch_f = b_o + b_1 a + b_2 d_f + b_3 d_t \tag{8}$$

where b_3 is the rate of change of hinge moment coefficient with flap deflection, d_t.

The function of the tab in balancing, servo control, and as a trimmer, is dealt with in the appropriate sections.

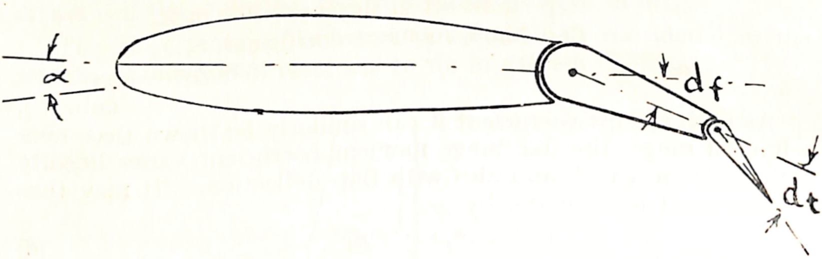


Fig. 5 (b).

A flap and tab arrangement.

The Control Force.

The relationship between control force and hinge moment may be determined, neglecting the work done in overcoming friction, by equating the work done *by* the pilot to the work done *at* the flap. Using the notation and geometry of Fig. 6 :—

$$\text{Work done by pilot} = \frac{P \times x}{2} \text{ in. lb.} \qquad \text{A.}$$

$$\text{Work done at flap} = \frac{H_f \times d_f}{2} \text{ in. lb.} \qquad \text{B.}$$

Equating A and B

$$\frac{Px}{2} = \frac{H_f d_f}{2}$$

$$\therefore \quad P = \frac{d_f}{x} H_f$$

$$\text{or } P = G \, . \, H_f \tag{9}$$

where G is the circuit gearing in radians per inch, that is the flap angular movement, in radians, per inch of control force movement.

i.e., Control Force = Gear Ratio × Hinge Moment (10)

Equation 10 shows that decreasing the gear ratio lowers the stick force for a given hinge moment. There is a limit, however,

to the minimum gearing possible governed by the maximum permissible control travel, the limit of flap travel being pre-determined by the magnitude of the moments required about the aircraft centre of gravity.

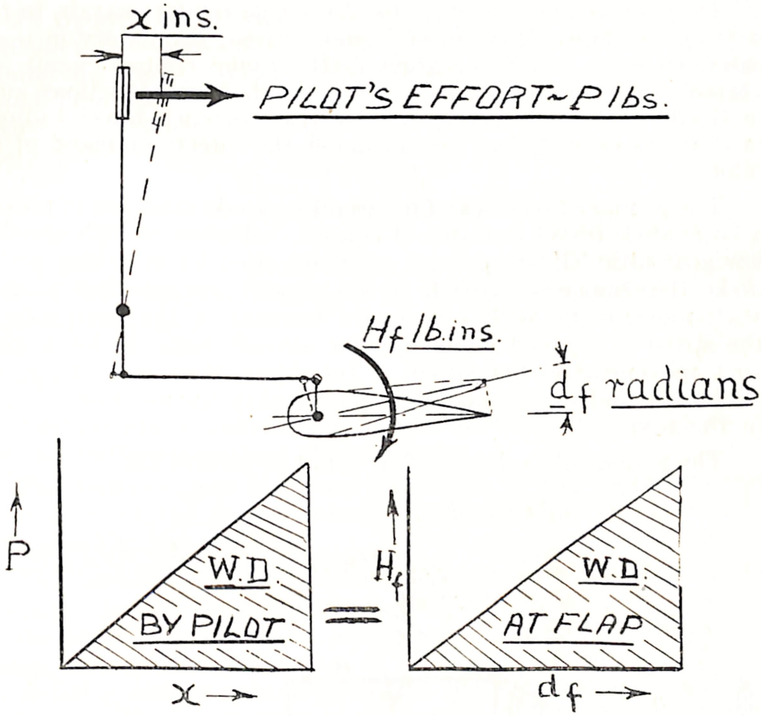


Fig. 6.

The relationship between control force and hinge moment.

Example 1.

The gear ratio of an elevator circuit is 3 degrees per inch.

(*a*) Find the maximum fore and aft travel of the stick grip if the elevator range is $\pm 21^\circ$.

(*b*) Determine the magnitude of the control force in terms of hinge moment.

(*a*) 3° deflection is produced by 1 inch of stick movement.

21° deflection is produced by 7 inches of stick movement.

Therefore the max. grip travel is 14 inches.

$$(b) \quad G = 3 \times \frac{2\pi}{360} = \frac{\pi}{60}$$

$$\therefore \quad \text{Control Force} = \frac{\pi}{60} \times \text{Hinge Moment—lbs.}$$

The gear ratio may not be fixed for a particular aircraft, in fact it is highly desirable in general, and, indeed, mandatory in many cases, to have variable gearing in the flying control circuit : a variable gear in the form of a spring tab which functions automatically and progressively with increase of stick force, and/or a mechanical gear change system under the direct command of the pilot.

The primary functions of a variable geared circuit are to provide a large stick travel in terms of control deflection at high speeds—low gear ratio 'G'—to prevent excessive stick forces arising and to make the change in attitude of the aircraft less sensitive to small stick movements, with progressive increase in the gear ratio as the speed is reduced giving greater aircraft response for a given stick movement at low speeds. Its object, then, is to defeat the stick force and speed squared proportionality law mentioned earlier in the text.

The spring tab is discussed in detail in Section IX.

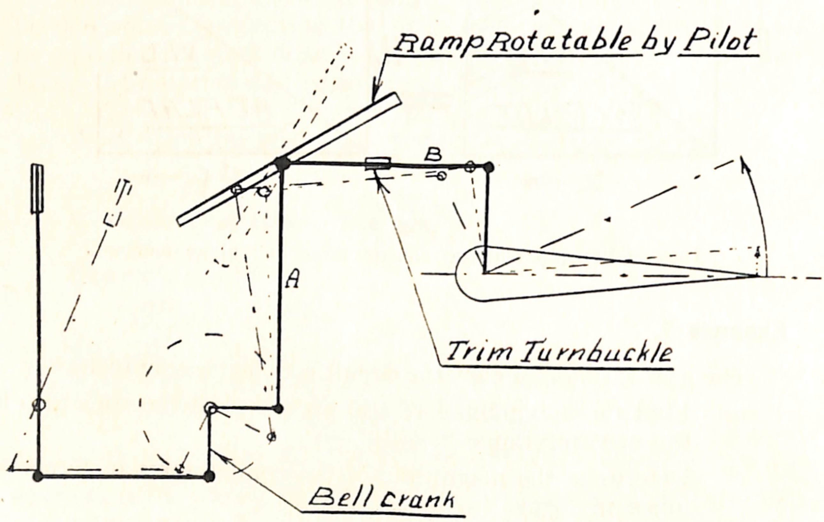


Fig. 7.

Schematic view of the Boulton Paul Variable Gear and Datum Change Control Mechanism.

A recently developed gear change system is the Boulton Paul variable gear and datum change control mechanism. In essence, the principle on which it is based is very simple and is shown in diagrammatic form in Fig. 7. The freely hinged joint connecting rods A and B is free to slide in ramp which may be rotated in flight by means of a handwheel located in the cockpit. Observation of the figure shows that increase in the angle of the ramp to the horizontal decreases the control surface travel in terms of control column movement. Thus, with increase in aircraft speed, the pilot would progressively enlarge the inclination of the ramp to the horizontal, and *vice versa*.

V. LATERAL CONTROL.

Lateral, or rolling, control can be achieved by the use of ailerons, spoilers, wing warping, rotating wing tips, elevons or differentially operated tailplanes.

Ailerons.

Ailerons are asymmetrically geared movable flaps situated at the rear outboard sections of the wings.

The *primary effect* of lateral movement of the control column is rotation of the aircraft about the longitudinal axis, namely, *rolling*. (Fig. 8).

The *secondary effects* of aileron deflection are *sideslipping* and *yawing*. In the banked attitude the inclination of the lift axis to the vertical produces an unbalanced component of lift which causes sideslip in the direction of the bank. Due to the sideslip, the relative airflow—the resultant of the forward speed and the sideslip velocity—is inclined to the longitudinal axis in the yawing plane to give the vertical tail surface an increased effective angle of attack. If the aircraft is correctly designed the increased aerodynamic force on the fin and rudder will produce a sufficient moment about the centre of gravity to yaw the aircraft in the direction of sideslip. This yawing tendency correcting sideslip is termed *weathercock stability*.

At low speeds the induced drag of the wings is relatively high and the moment about the centre of gravity caused by the aileron drag may overcome the weathercock stability to give yaw away from the direction of sideslip, namely, adverse yaw. (See also section on adverse yawing moment of ailerons).

The basic function of ailerons is to provide a rolling moment with little nett change in lift. Lateral movement of the control column, say, to the port (Fig. 8) produces a downward deflection

of the starboard aileron with a simultaneous upward deflection of the port aileron.

The aileron deflected downwards creates an increase in lift whilst on the other wing the upward deflection causes a decrease in lift. This asymmetric lift distribution induces a rolling moment accelerating the aircraft in roll.

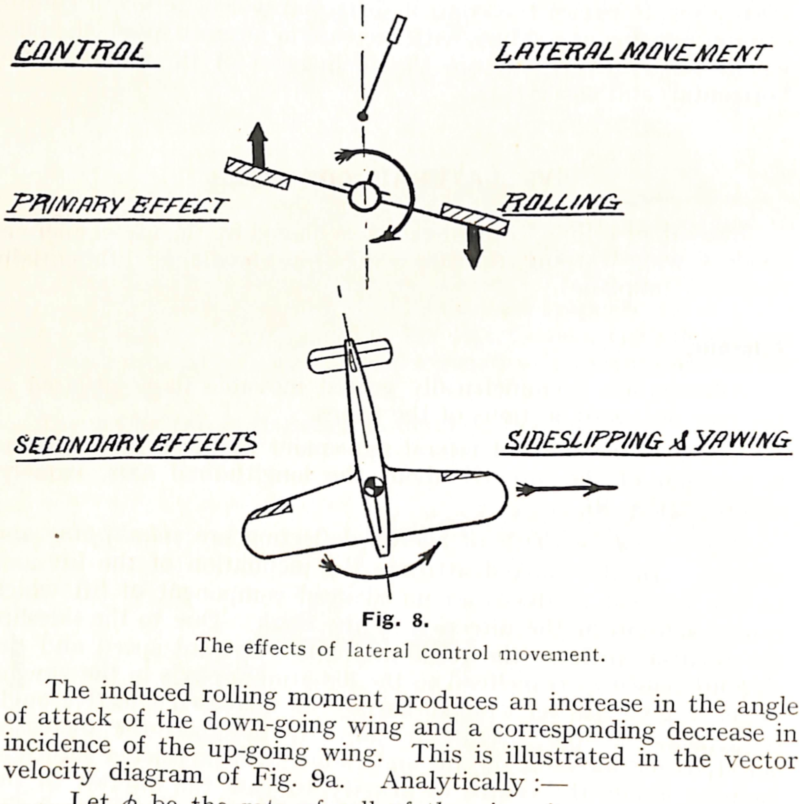


Fig. 8.
The effects of lateral control movement.

The induced rolling moment produces an increase in the angle of attack of the down-going wing and a corresponding decrease in incidence of the up-going wing. This is illustrated in the vector velocity diagram of Fig. 9a. Analytically :—

Let p be the rate of roll of the aircraft.

Let b be the semi-span of the wings.

and V be the true airspeed of the aircraft.

Then the resultant velocity of the airflow at the wing tip is given in magnitude and direction by the velocity vector V_R.

The vector triangles of velocities give that :—

The tangent of the change in attack angle $= \pm pb/V$,

which approximates to :—

The change in wing tip angle of attack produced by roll

$$= \pm pb/V.$$

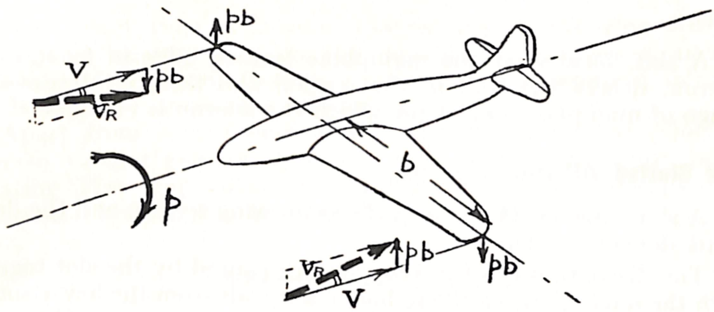


Fig. 9 (a).
The change in effective angle of attack due to roll.

If the aircraft is *not near its stalling attitude* the increase (decrease) in angle of attack produces more (less) lift. (Fig. 9b). This sets up a moment resisting the roll ; a damping moment of the opposite sense to that produced by the deflected ailerons.

When the damping moment becomes equal to the applied rolling moment a steady rate of roll is achieved. That is :—

Applied Rolling Moment = Damping Rolling Moment.

For a particular aileron deflection :—

The applied rolling moment α V_e^2
and the damping rolling moment α $V_e^2 \times p/V$
thus p α V, or, in other words :—

The steady rate of roll α true airspeed (for a given aileron angle)
(α is the sign of proportionality). 11.

If, however, the aircraft *is near the stalling attitude* an increase in the angle of attack may cause a loss of lift. Fig. 9c illustrates this remark.

It explains, too, the ineffectiveness of the aileron at the stall.

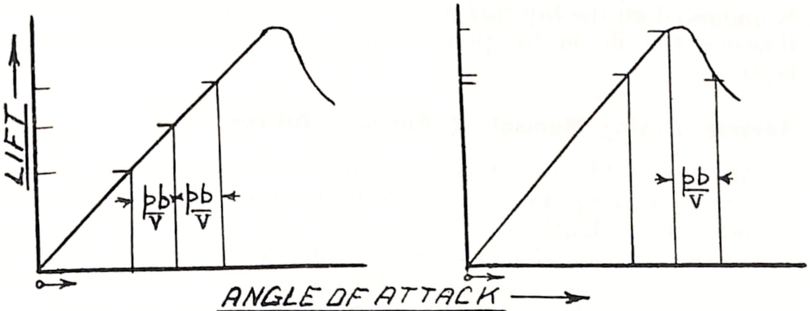


Fig. 9 (b). **Fig. 9 (c).**

The effect on lift due to change in angle of attack.
(a) Below stalling angle. (b) Close to stalling angle.

Slats.

A slat, located at the mainplane leading edge in front of the aileron, delays the stall in that region and thereby increases the range of mainplane angles for which the aileron is effective.

The Slotted Aileron (Fig. 10).

A slot is provided between the main wing section and the downward deflected aileron.

The discontinuity of the top surface caused by the slot together with the ducting of relatively high energy air from the lower surface through the slot to the upper surface provides a measure of boundary layer control which delays separation of the flow over the rear portion of the wing.

Stalling of the wing due to aileron depression is delayed and more powerful rolling moments are achieved at greater downward aileron deflections.

Slotted ailerons are used on the carrier based Supermarine Attacker.

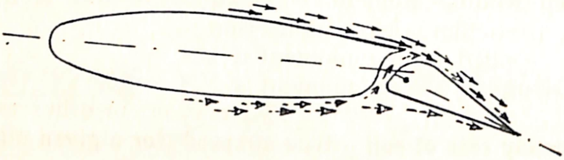


Fig. 10.

The slotted aileron.

The Boundary Layer Fence.

A boundary layer fence is a shallow chordal fence which may be mounted on the top surface of swept back wings to improve the flow over the ailerons by opposing the spanwise drift in the boundary layer.

Adverse Yawing Moment of Ailerons—Aileron Drag.

An undesirable secondary effect due to the deflection of *simple* ailerons is to cause an adverse yawing moment. It is sometimes termed aileron drag. This drag effect was touched upon when considering the drag due to a simple flap deflection. On the wing with the down-going aileron, lift and hence induced drag is increased (for the induced drag coefficient is proportional to the square of the lift coefficient). Also profile drag is increased. On the wing with the up-going aileron profile drag is also increased but the effect of the decrease in lift is to cause a reduction in induced drag.

The nett effect, then, is to cause a larger drag on the wing with the down-going aileron than on the one with the upgoing aileron. This initiates a yawing moment contrary to that required, in other words, an adverse yawing moment.

Apart from synchronous rudder movement by the pilot the adverse yawing moment tendency can be neutralised by incorporating Frise type ailerons; by providing the ailerons with differential gearing; or by a combination of these two.

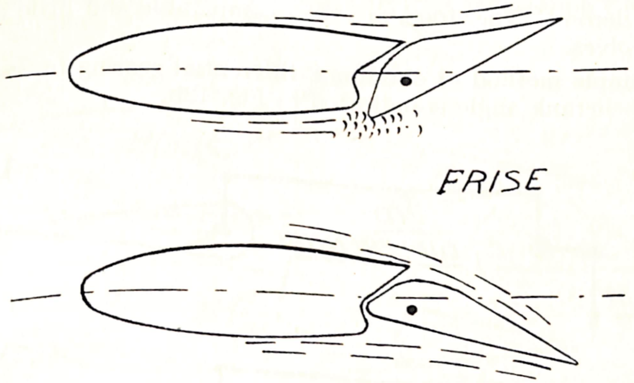


Fig. 11.
The Frise Aileron.

Frise Ailerons (Fig. 11).

Frise ailerons are ailerons that are specially shaped at the nose to ensure that the nose of the up-going aileron projects into the airstream to increase the profile drag and attentuate the effect of the decrease in induced drag relative to the down-going aileron with the screened nose on the other wing.

This ensures that the drags on each wing are comparable, mitigating the aileron drag effect.

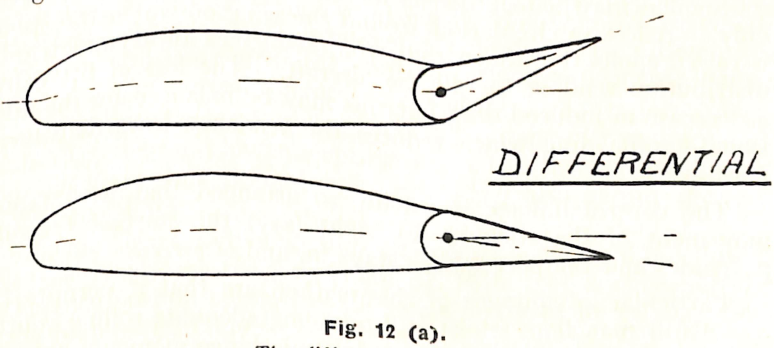


Fig. 12 (a).
The differential aileron.

Differential Ailerons (Fig. 12a).

Differentially geared ailerons are those which for the same amount of stick movement each side of neutral the control surfaces move further in one direction than the other.

Differential gearing minimises the adverse yawing effect if the up-going aileron moves through a larger angle relative to the down-going one. A suitable gear ratio gives additional profile drag on the up-going to compensate for the extra induced drag on the down-going aileron. The drags are then comparable and little adverse yaw evolves.

A simple method of obtaining differential gearing by variation of the bellcrank angle is indicated in Fig. 12b.

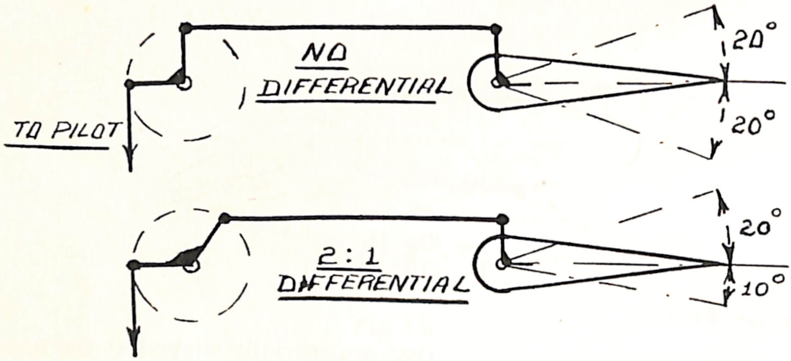


Fig. 12 (b).

The achievement of differential gearing by variation of the bellcrank angle.

Spoilers (Fig. 13a).

Spoilers may be regarded as ailerons which move in one direction only. A loss in circulation around the wing on which the spoiler is raised spoils the lift on that wing and gives an asymmetric lift distribution tending to roll the aircraft. The loss of lift causes a decrease in induced drag but this may be balanced by the additional profile drag which reduces the tendency towards adverse yawing.

The control linkage is usually so arranged that with lateral movement of the stick to the right (say) the starboard spoiler protrudes and the port spoiler stays in, and *vice versa*.

Particular advantages of this method are that it permits the use of full span flaps whilst still retaining adequate rolling control and also that the associated aerodynamic hinge moments are low.

A disadvantage is the slower response to control change. Positioning of the spoiler further aft quickens the aircraft response but only at the expense of control effectiveness. Probably the greatest objection to the use of the spoiler is that the effect is always to reduce the lift. This is of particular importance at low altitude as, of course, coupled with the decrease in lift is a consequent loss of height.

These limitations tend to restrict the use of the spoiler as the sole rolling control and its chief utilisation in this respect is probably as an aileron augmentor. It is used for this purpose on the Lockheed Neptune.

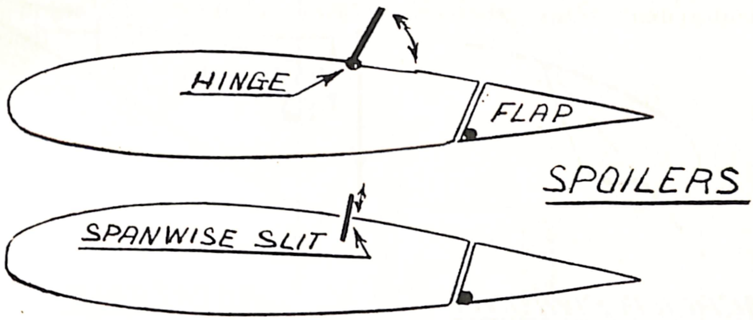


Fig. 13 (a).
Spoiler arrangements.

Wing Warping.

Wing warping was an early but effective method of obtaining rolling control by aerodynamic means. It was used with success on the Wilbur and Orville Wright biplane of 1903.

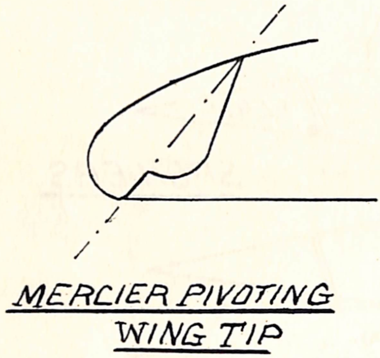
Rotating Wing Tips.

The modern modified version of wing warp, rotating wing tips can be thought of as ailerons—or elevons—whose chords are equal to the wing chord. Rolling moment is produced by differential rotation of the wing tips, the control surface movements being characteristic of normal ailerons or elevons.

In high speed aircraft a particular advantage of rotating wing tips is that they are less susceptible to control reversal (see later) than the more conventional ailerons. They also permit the use of long span flaps.

Fig. 13b shows a pivoting wing tip having a diagonal hinge line. This type is used on the Aerosudest SE 1010. It is claimed that the balance portion forward of the hinge increases its effectiveness during high speed flight.

The rotating wing tip illustrated in Fig. 13c is similar to that used in the Short Sherpa. In this aircraft the tips can move in phase as 'elevators' and asymmetrically as 'ailerons.' The rotating wing tip is completely balanced aerodynamically when the axis of rotation coincides with the centre of pressure line—about $\frac{1}{4}$ chord for subsonic flight. This occurs in the Sherpa, the correct degree of partial balance being achieved by the use of anti-balance tabs.



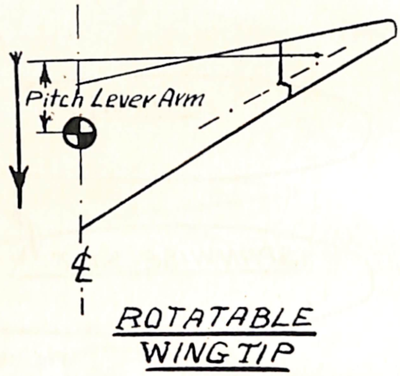


Fig. 13 (b). Fig. 13 (c).

Variable incidence wing tip controls.

Elevons.

The name is very aptly derived from ELEVator and ailerON as the elevon performs dual functions, the functions of both elevator and aileron.

For lateral stick movement the elevons move asymmetrically as 'ailerons' whereas for fore and aft movement they move in phase as 'elevators.'

Favoured, particularly in the United States of America, as an attempt to reduce the number of control surfaces on delta aircraft (Fig. 14) at the price of loss in efficiency as 'ailerons' at large 'elevator' deflections.

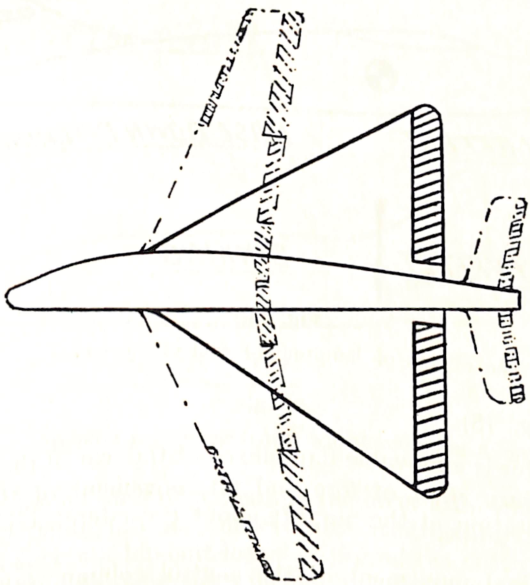
The Differentially Operated Tailplane.

A recent innovation, this unorthodox means of providing lateral control, renders the aircraft completely controllable by its tail.

Used on the supersonic S.O. 9,000 Trident, the whole tailplane is movable by the pilot both asymmetrically as 'ailerons' and in unison as 'elevators.'

A disadvantage would appear to be the shortness of the rolling lever arm, although this could be offset by a larger control surface area, and also by the more powerful characteristics of a movable main surface.

The advantages of this type of rolling control are probably peculiar to the Trident, rather than general, in that its wings, which are of short span, have launching turbo-jets mounted in pod form at the tips. This indicates that the alternative positioning of the lateral control is probably a secondary rather than primary design consideration.



ELEVONS

Fig. 14.

The reduction in number of control surfaces by use of elevons.

VI. LONGITUDINAL CONTROL.

Longitudinal control, or pitching control, is usually achieved by elevators but may also be obtained by elevons, adjustable tailplanes (flying tails) and butterfly tail arrangements.

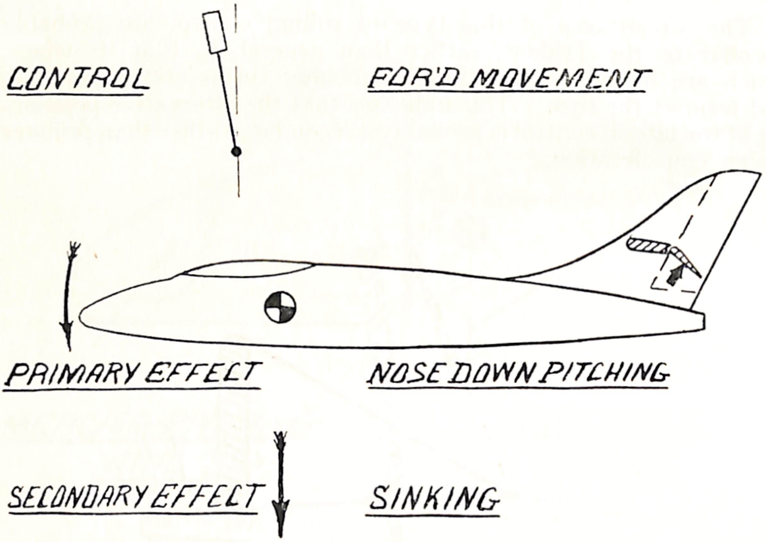


Fig. 15.

The effects of longitudinal control movement.

Elevator (Fig. 15).

An elevator is a movable flap hinged at the rear of the tailplane.

The *primary effect* of fore and aft movement of the control column is rotation of the aircraft about the lateral axis, namely, *pitching*.

Longitudinal movement of the control column, say, forward causes a downward deflection of the elevator. The consequent increment of lift on the horizontal tail surface produces an anti-clockwise moment about the centre of gravity of the aircraft, the primary effect on the aircraft thus being a nose down pitching tendency.

The *secondary effects* of elevator deflection are *sinking with increase in airspeed* in the case of *nose down pitch,* and *rising with decrease in airspeed* with *nose up pitching motion.*

The pitching moment produced is dependent on the magnitude of the resultant aerodynamic force induced on the horizontal tail

surface area by the deflected elevator and also on the perpendicular distance of its line of action from the centre of gravity of the aircraft.

The longitudinal disposition of the horizontal tail relative to the centre of gravity thus plays a large part in the determination of the magnitude of the induced pitching moment. As the aerodynamic force itself is proportional to the plan area of the tail surface, then, a particular pitching moment can be obtained either with a short lever arm with large horizontal tail area or a long lever arm with a small horizontal area.

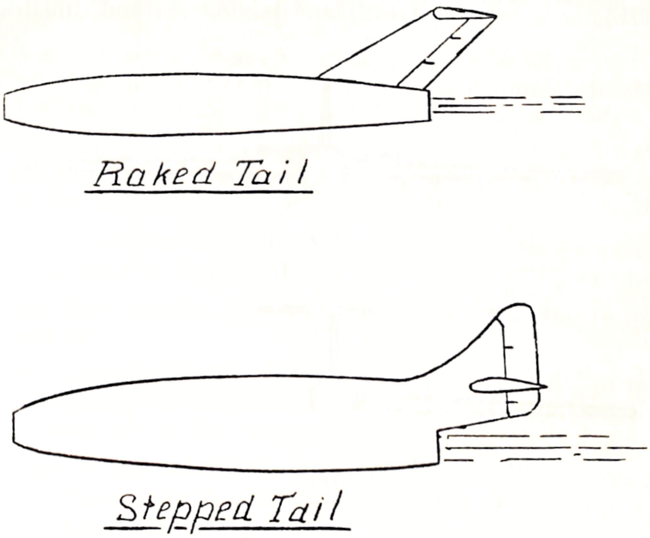


Fig. 16.

Methods of obtaining a long lever arm whilst retaining a short jet pipe.

With some jet aircraft increasing the lever arm, whilst retaining simple tail configurations, necessitates lengthening the jet pipe often with severe attendant losses in propulsive efficiency. Alternative solutions in these cases include the incorporation of a raked tail as in the MiG 15 or a stepped rear fuselage similar to that used in the SAAB J29. These configurations are illustrated in Fig. 16.

The vertical disposition of the horizontal tail surface is also of prime importance.

Fig. 17a shows a typical elevator location on a low speed piston-prop aircraft.

Figs. 17b, 17c and 17d show more likely positions for high speed jet aircraft. The higher location is necessary mainly to keep it clear of the wake of the wings, canopy, flaps and jets. The wing and

canopy wake case is usually particularly critical in the transonic zone as it could institute tail buffeting and reduction in control efficiency ; whereas, the effect of flap wake is most critical at high angles of attack, for instance, when landing. The tail unit has to be kept out of the jet wake for two reasons :—

1. To avoid the possibility of the hot jet stream from impinging on the horizontal tail.
2. To prevent the velocity effects of the jet stream from affecting the efficiency of the control surface.

The high setting may be obtained by a high tee tail configuration (Fig. 17b), or a lower set tailplane having marked dihedral (Fig. 17c).

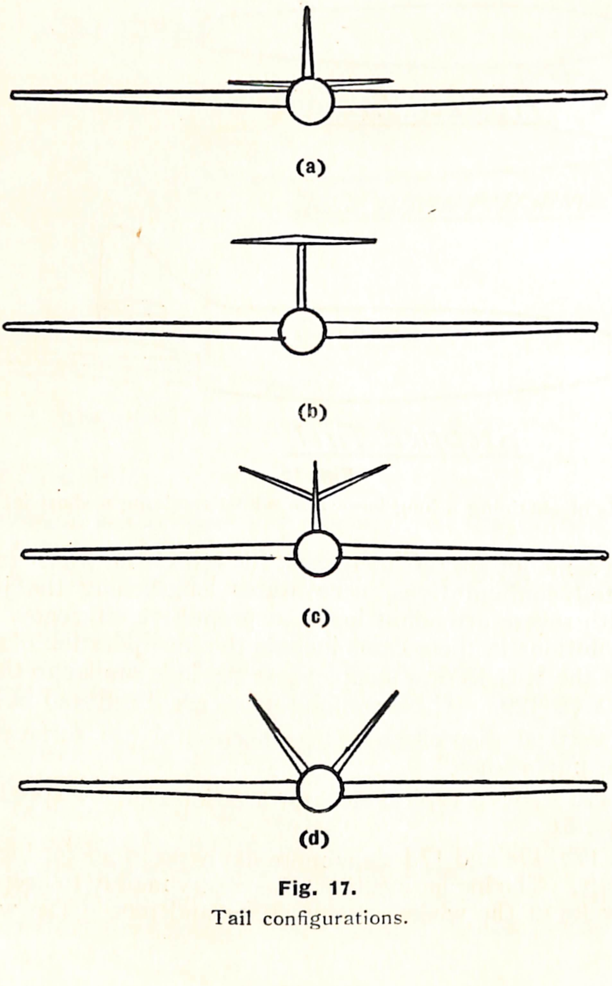


Fig. 17.

Tail configurations.

The proximity of fuselage, horizontal tail and vertical tail on aircraft designed to fly at high subsonic speeds must also be so arranged to avoid coincident peak pressure positions, and thus peak local Mach numbers, Mach numbers greater than the free flight Mach number.

Disturbances causing these peak local velocities, that is, giving the aircraft the 'Machs', are often elusive to locate and ensuant modifications may well chase the Machs elsewhere.

The designer keeps many cards up his sleeve to play on meeting roughness, shake, etc., during high subsonic flight tests, to improve the flow characteristics over the tail unit. These include the use of :—

1. Bullet, tapered or acorn fairings on the front or rear of the fin tailplane junction to obtain better three dimensional flow than with the more conventional hollow fillets.
2. Horizontal separating plates on the fin leading edge to prevent upflow and separation of flow on the upper surface of the horizontal tail unit.
3. A dorsal fin to move the peak suctions on the fin away from those on the horizontal tail.
4. Air flow strakes (spray strips) along the fin to prevent upflow.
5. A longitudinal spine connecting canopy and fin to improve the quality of flow over the tail unit by reducing the turbulence aft of the canopy.

Equation 4 and subsequent remarks showed the relative powerfulness of main section and flap deflection (*e.g.*, tailplane and elevator deflections).

With supersonic flow conditions existing around the tailplane the relative power is further accentuated in two ways.

As was shown in a general way in Section IV the effect of elevator deflection, for low subsonic flight, is to change the pressure distribution and hence the lift on both elevator and tailplane. If, however, the aircraft speed is such—and not necessarily supersonic —as to produce shock waves on the upper and lower surfaces of the tailplane, then, as is characteristic of shock front conditions, elevator movement will little influence the flow forward of the shock waves.

Thus only infinitesimal lift changes forward of the shocks are associated with elevator deflections when these conditions exist.

Due to the break-away of flow aft of the wave fronts the range of elevator movement may well lie within a stagnant air region. This is illustrated in Fig. 18.

Thus only small lift changes aft of the shocks are associated with such elevator deflections.

Under these conditions, then, it is seen that elevator effectiveness is practically non-existent.

The onset of these conditions may be sufficiently delayed by using a thin aerofoil section for the horizontal tail, and by reducing the effective velocities—the velocity components perpendicular to the leading edge—over the surface by sweepback of the leading edge.

More drastic measures include dispensing with the horizontal tail surface, pitching control being then achieved by rotating wing tips; and by the use of an all-moving horizontal tail.

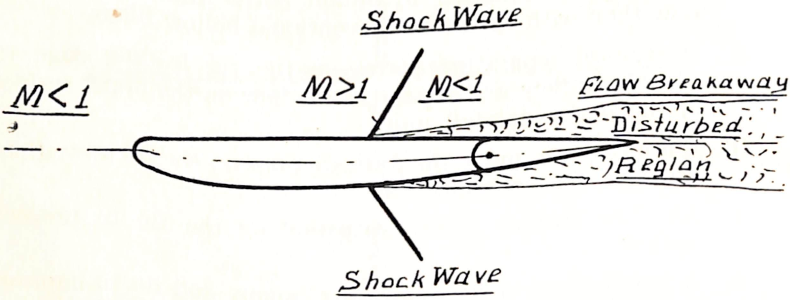


Fig. 18.
The breakaway of flow and consequent wake spread with onset of shock stall.

The Slotted Elevator.

A slot is provided which permits free passage of air between the trailing edge of the tailplane and the leading edge of the elevator when the elevator is upwardly deflected.

The discontinuity of the lower surface caused by the slot, together with the ducting of relatively high energy air from the upper surface through the slot to the lower surface provides a measure of boundary layer control which delays separation of the flow over the rear portion of the horizontal tail surface.

'Inverted' stalling of the horizontal tail surface due to upwardly deflected elevator is delayed and more powerful nose-up pitching moments are possible at greater elevator deflections.

Inverted Slats.

The inverted slat located at the leading edge of the tailplane is another horizontal tail surface anti-stall device.

It is used on the high wing Prestwick Pioneer to prevent inverted stall of the horizontal tail surface with the nose down change of trim associated with the mainplane flap and slat retraction.

Elevons.

The function of the elevon as an 'elevator' has already been discussed under "Rolling Controls." A particular disadvantage is the shortness of the pitching lever arm. This is shown in Fig. 13c.

Adjustable Tailplanes.

Used in early aircraft and again coming into vogue—but for different reasons—the 'flying tail' consists of an all moving horizontal tail which acts as an elevator.

Apart from greater effectiveness than the conventional elevator under high subsonic flight conditions corresponding with the shock stall and associated break-away of flow, as shown in Fig. 18, its use tends to decrease the likelihood of control reversal. This will be dealt with at a later stage. A constructional disadvantage is that some difficulty is experienced in providing efficient root sealing.

Vee Type or Butterfly Tail.

The Vee tail was first introduced by a Polish engineer, Rudlicki, in the early 30s.

Illustrated in Fig. 17d the movable surface of the butterfly tail can be operated in phase as an 'elevator' and asymmetrically as a 'rudder.' Its use decreases the number of control surfaces and eliminates the possibility of peak suctions occurring at vertical and horizontal tail surface junctions.

This type of tail is further referred to in the section on "Directional Control."

VII. DIRECTIONAL CONTROL.

Directional, or yawing, control is generally effected by a rudder, but occasionally it is achieved by use of a Vee type tail arrangement, or movable fin.

Rudder.

The rudder, a movable flap hinged to the rear of the fixed vertical tail surface (the fin), effects yawing control.

The *primary effect* of rudder control movement is rotation of the aircraft about the normal axis, namely, *yawing*.

With, say, right foot forward and left foot aft movement on the rudder bar the rudder is deflected to the starboard, as shown in Fig. 19. The consequent moment produced about the aircraft centre of gravity by reason of the change in aerodynamic force—'lift' normal to the lifting plane—on the vertical tail surface causes the primary effect of rudder movement, a yawing tendency to starboard.

The *secondary effects* of rudder deflection are *crabbing* and *rolling*.

The tendency of the aircraft to continue along its original line of flight when yawing occurs brings about a resultant motion known as crabbing. Also the increase in lift occurring on the faster moving outer wing together with the decrease in lift on the slower moving inner wing, during the yaw, initiates a rolling motion in the direction of the crabbing.

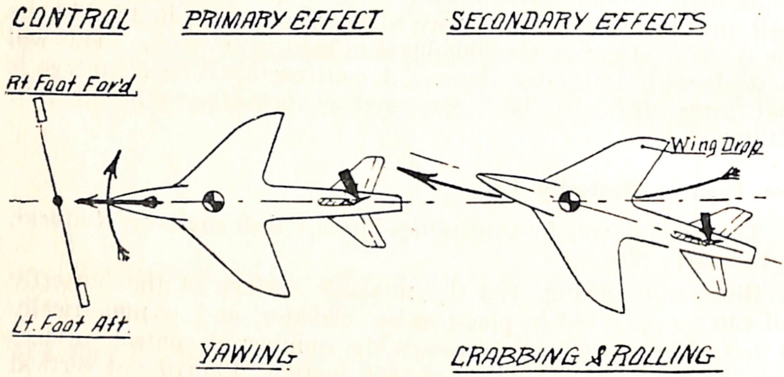


Fig. 19.

The effects of directional control movement.

Vertical Tail Stalling and Rudder Locking.

If the angle of sideslip becomes sufficiently large—about 15°—stalling of the fin and rudder combination occurs. The consequent movement aft of the centre of pressure may cause a reversal of rudder hinge moment causing the rudder to float over and aggravate the yaw and hence also the stall.

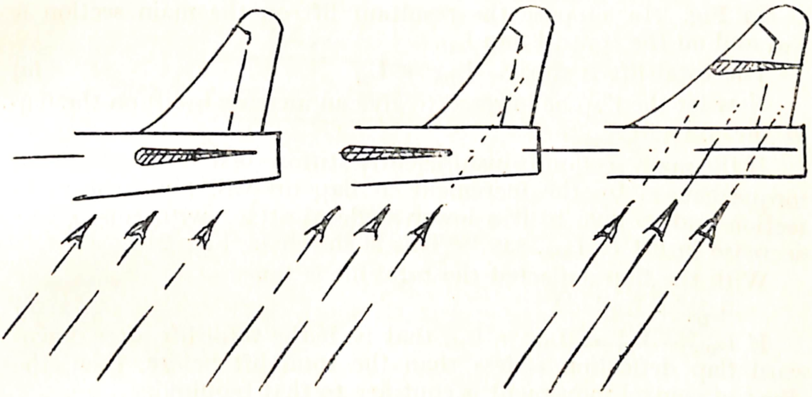
In severe cases the increase in rudder angle may be sufficient to cause it to 'jam over' and possibly become immovable by the pilot, to produce a particularly dangerous condition known as rudder locking.

If an aircraft has a tendency towards rudder locking it may be overcome by the installation of a dorsal fin, that is, a forward extended fin tapered in side elevation with a long root fillet, as shown in Fig. 15.

Spin Characteristics.

The proximity of horizontal tail surface and rudder is of importance when considering the spin characteristics of an aeroplane.

In a spin an aircraft is in a highly stalled condition. That is, the angles of attack of both wings and horizontal tail are greater than the stalling angle, and so both aileron and elevator controls may be ineffective. This indicates that the rudder may be the only effective control, the only means of recovery, in a spin. It is thus necessary to ensure that the rudder is not blanketed by the horizontal tail surface under spin conditions as this might render it ineffective.



(a) **(b)** **(c)**

Fig. 20.

The effect of horizontal tail surface location on spin characteristics.

Fig. 20 shows some relative horizontal and vertical tail surface locations.

(*a*) Is completely shielded and thus unsatisfactory.

(*b*) Shows the beneficial effect of forward displacement of the horizontal tail surface. A similar effect could be obtained by aft displacement as in the Supermarine Attacker.

Another satisfactory combination, a mid-high tailplane location —as in the Avro Canada CF 100—is the type illustrated in (c).

A more rigorous application of this approach is the Tee type tail, as in the Gloster Javelin, and shown diagrammatically in Fig. 17b.

Two other satisfactory tail arrangements from the point of view of rudder effectiveness under spin conditions are the butterfly tail (Fig. 17d), and the use of twin vertical tail surfaces mounted at the ends of the horizontal tail. Aircraft incorporating butterfly

tails include the Beechcraft Bonanza and the Supermarine Type 508 ; whereas a twin rudder example is the de Havilland D.H. 110.

Movable Fin.

With this arrangement the entire vertical tail surface, the fin, moves to effect directional control (*e.g.*, S.O. 9000 Trident).

VIII. CONTROL REVERSAL.

In Fig. 21a suppose the resultant lift on the main section is L_{1m} and on the control flap L_{1f}.

The total lift is thus :—$L_{1m} + L_{1f}$ (*a*)

Now let the flap be lowered to give an increase in lift on the flap to the amount L_{2f}.

If the main section is insufficiently stiff in torsion the additional torque caused by the increment in flap lift will twist the main section and so give to it a lower angle of attack with consequent decrease in lift to L_{2m}, say. This is shown in Fig. 21b.

With the flap deflected the total lift is thus :—

$L_{2m} + L_{2f}$ (*b*)

If $L_{2m} + L_{2f} < L_{1m} + L_{1f}$, that is, if the total lift after downward flap deflection is less than the total lift before, then, the effect of control movement is contrary to that required.

This phenomenon is known as 'Control Reversal.'

Summarising, control reversal occurs when the loss (gain) in lift due to main section twist is more than the gain (loss) in lift due to positive (negative) flap deflection.

The main surface and flap being the wing and aileron, respectively, in the case of roll ; and the tailplane and elevator in the case of pitch.

For an aileron, control reversal is highly undesirable ; for an elevator, control reversal can well be catastrophic if, say, the elevator is being used for a 'pull-out' from a dive.

It has been shown that for a rigid wing and a given aileron deflection the rate of roll of an aircraft is proportional to its true airspeed (Equation 11).

If, however, at a particular speed twisting of the wing commences and increases with the speed then the rate of roll will decrease until at the 'Aileron Reversal Speed' the rate of roll disappears. This is denoted by the speed V_{rs} on the graph in Fig. 21*c*.

Steps taken at the design stage to ensure that the reversal speed is beyond the maximum speed of the aircraft include :—

1. Increase of the wing, or tailplane, torsional stiffness.
2. Incorporation of rotating wing tip controls in the case of roll and 'flying tails' in the case of pitch.

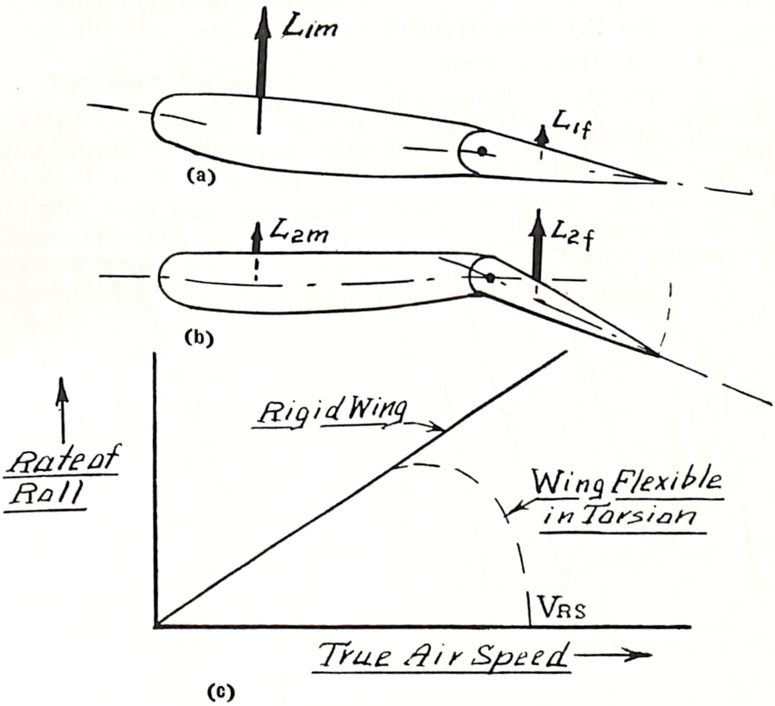


Fig. 21.

(a), (b) The twisting of a main section flexible in torsion due to the torque produced by flap lift increment; (c) The aileron reversal speed.

IX. CONTROL BALANCE.

There are two general forms of control balance : one is concerned with control force reduction ; the other with flutter prevention.

The primary object of *aerodynamic, mechanical and tab balance* is *to reduce the effort required of the pilot to manipulate the basic flying controls.*

Whereas, the primary object of *mass balance* is *to prevent flutter.*

A. **Aerodynamic, Tab and Mechanical Balance.**

Aerodynamic Balance.

Aerodynamic balance attacks the problem at the root, its objective being reduction in control surface hinge moments, which, consequently, ensures lighter controls.

The principal methods of aerodynamic balance are :—

1. The Set Back Hinge Balance.
2. The Horn Balance.
3. The Sealed Internal Pressure Balance.
4. Special Profile Shapes.

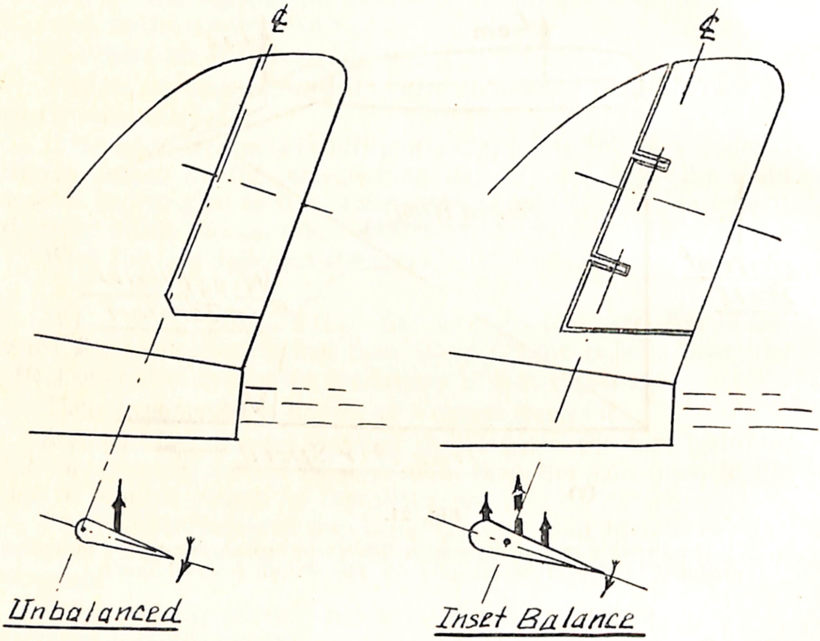


Fig. 22.

The effect of inset hinge: The pressure for'd of the hinge brings the c.p. close to the hinge axis.

1. The Set Back Hinge (Inset Balance).

The set back hinge type balance is one in which the control surface is hinged some distance aft of the leading edge. It is in general use, often in conjunction with the geared tab. Fig. 22 shows a diagrammatic view of this balance and compares it with an unbalanced control surface.

The effect of aft displacement of the hinge is to bring the line of action of the resultant aerodynamic force on the control surface (dashed line in Fig.) nearer the hinge line. This reduces the hinge moment and hence the stick force for a given control movement and condition similitude. Or, using another argument, the effect of the aerodynamic force for'd. of the hinge line is to cause a moment about the hinge axis, a moment assisting control deflection.

2. The Horn Balance.

The horn balance is similar to the set back hinge in principle and effect, but different in geometry, in that the protuberance for'd. of the hinge is local, rather than distributed.

This local protuberance is termed the horn. If the horn is completely screened by the main surface it is termed a shielded horn, if semi-screened a partly shielded horn, and unscreened an unshielded horn. These types are illustrated in Fig. 23.

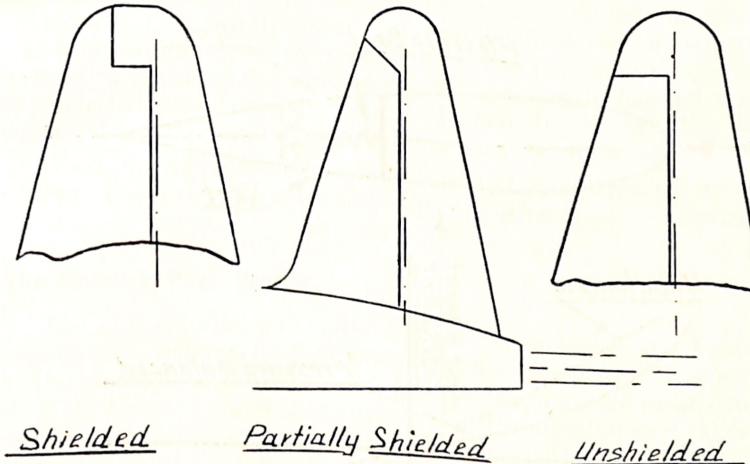


Fig. 23.
The horn balance.

3. The Internally Sealed Pressure Balance (Fig. 24).

Although of pre-war design, by Irving, and developed at the National Physical Laboratory, not until recently has it been put into service in British aircraft. It is now used on the control surfaces of the Vickers Valiant and English Electric Canberra, etc.

Preventing gap leakage in itself greatly increases the efficiency of a control surface. The sealed pressure balance accomplishes this by way of a flexible seal and also reduces the hinge moments

by utilising the pressure differential at the vents to produce a moment assisting the pilot. As the control surface is lowered (say) the pressure reduces in the vicinity of the top vent and increases around the lower vent. This pressure differential existing between bottom and top surfaces of the tongue sets up a moment about the hinge assisting control deflection, that is, a balancing moment.

The plot of hinge moment coefficient and control deflection for the pressure balance is seen to be almost linear for a considerable range, as with the idealized simple control, but of reduced slope. This indicates that the hinge moment and hence the control force is reduced and that this partial balance is maintained for appreciable control deflections.

This is a type of balance which is reasonably insensitive to rigging and one with a great potential in view of its efficiency at high aircraft speeds. Design and upkeep problems are increased, however, by its relatively complex construction.

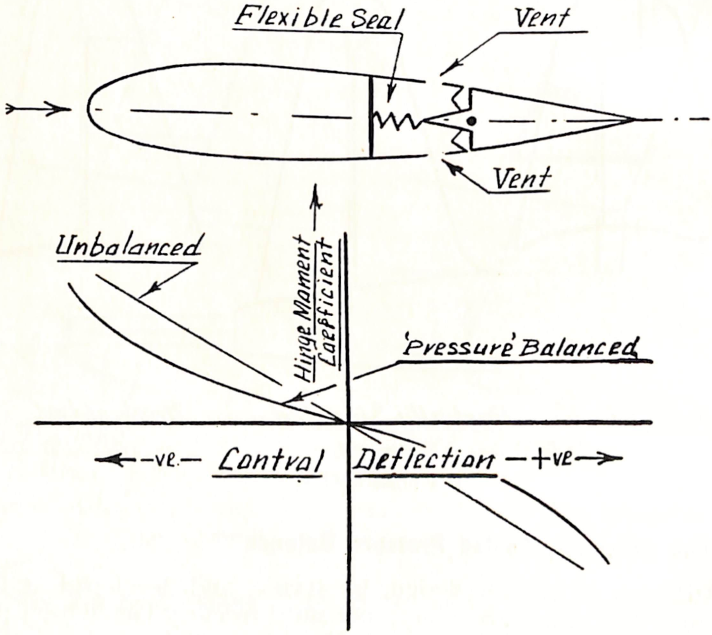


Fig. 24.

The sealed internal pressure balance:—comparison of unbalanced and 'pressure' balanced hinge moment coefficients.

4. Special Profile Shapes—The Frise and Handley Page Profile—The Bevelled Trailing Edge.

The Frise Type Profile (Fig. 11).

Described and illustrated in the section on rolling control to show its effect on yawing characteristics, the Frise profile may also be used on ailerons to achieve a degree of aerodynamic balance.

The effect of nose protrusion into the airstream is the promotion of an aerodynamic force on the control leading edge which sets up a moment about the aileron hinge axis aiding control deflection.

The curve of Fig. 25c shows that the relation between hinge moment coefficient and aileron deflection is far from linear. This indicates that the control surface rigging is critical and that there is danger of overbalancing under certain conditions. Indeed, in this case, it is seen that for small upgoing angles the rate of change of hinge moment coefficient with aileron deflection (*i.e.*, b_2) is $+ve$. Thus for this range the aerodynamic force on the surface is such as to produce a sufficient moment to increase the control angle and a *reversal of stick force* occurs, that is, the aileron is overbalanced. Fortunately there are two ailerons on an aeroplane and this instability, or overbalance, on the part of one aileron is utilised in assisting the movement of the opposite handed stable, or underbalanced, aileron. Thus, if correctly rigged, the right degree of partial aerodynamic balance may be obtained.

Up-rigging of both ailerons may tend to overbalance the controls whereas down-rigging may have the opposite effect of increasing the control forces.

The Handley Page Profile.

The H.P. profile, as shown in Fig. 25a, achieves its degree of aerodynamic balance by the same method as the Frise. In contrast to the unbalanced control, which has the leading edge radius struck from the hinge axis and thus ensures that the aerodynamic pressures on the nose (which are perpendicular to the surface) do not generate moments about the hinge line, the balanced nose of the H.P. profile has a symmetrically tapered leading edge well for'd. of the hinge axis.

The Bevelled Trailing Edge (Fig. 25b).

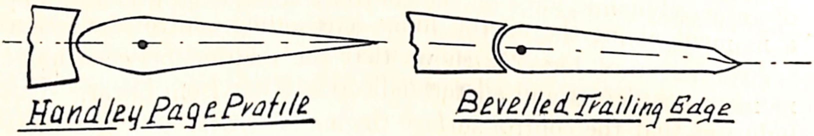
This type makes use of the effect of trailing edge angle on balance characteristics. A United States method of approach to the problem and one that has not found great favour in this country, the British opinion being in general diametrically opposed to this in that a control trailing edge is considered to be the place to fit bulbous 'bite' devices and anti-snaking strips, etc.

Tab Balance.

This form of balance is often classified under the general title of aerodynamic balance. It may take the form of :—

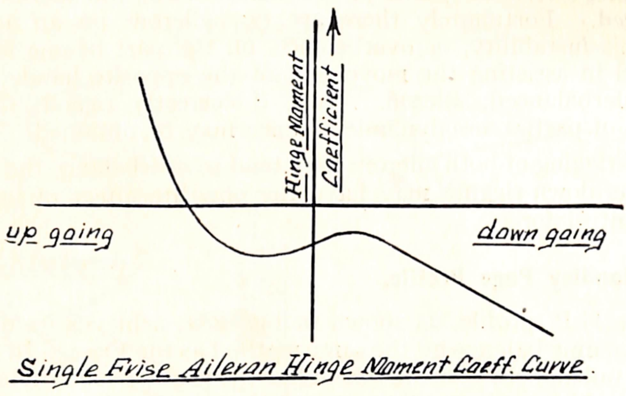
(*a*) The Geared Balance Tab.

or (*b*) The Spring Tab.



(a) **(b)**

Aerodynamic balance by special profile shapes.



(c)

Fig. 25.

The non-linear single Frise aileron hinge moment coefficient versus deflection curve.

(*a*) **The Geared Balance Tab** (Fig. 26).

A geared balance tab is an auxiliary flap inset in the trailing edge of the main control surface and so linked that as the movable main surface is deflected by the pilot the tab moves in the opposite direction. This results in a main control hinge moment reduction of an amount equal to the product of the resultant aerodynamic

force on the tab and the distance of its line of action from the main control hinge axis.

Complete balance could be achieved if the effect on hinge moment coefficient of tab deflection is made equal and opposite to that of the corresponding movement of the main control surface. Referring to equation 8, this may be expressed mathematically as :—

For total balance $b_2 d_f = -b_3 d_t$

$$i.e., \frac{d_t}{d_f} = -\frac{b_2}{b_3} = \text{a constant} = \text{Gear Ratio} \qquad (12)$$

Thus, within the range for which the tab is effective, increase in gear ratio also increases the degree of balance. A mechanical means of changing the gear ratio is shown in Fig. 26, which figure also illustrates its effect on the hinge moment coefficient.

The upper limit of partial balance that can be practically achieved is governed by ensuring that overbalance is impossible under the most adverse conditions of manufacture, and service, and also by the ineffectiveness of the tab at large deflections.

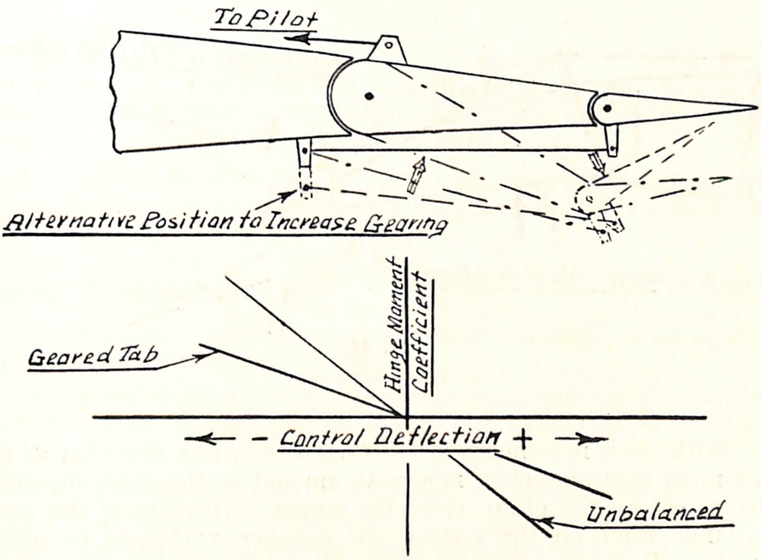


Fig. 26.

The geared balance tab.

(*b*) **The Spring Tab.**

The spring tab is designed to 'beat' the stick force and square of the airspeed proportionality law. It may be thought of either as a variable geared balance tab, the gearing being automatically increased with greater aircraft speed so as to reduce the dependency of stick force on speed ; or even as a geared tab-cum-servo. A spring with a lesser rate (*i.e.*, more elastic) would increase the gearing, the limit being, of course, when the spring has infinite elasticity, that is, if it were removed, when the system would act as a pure servo tab arrangement.

The modus operandi is simple. Fig. 27 refers.

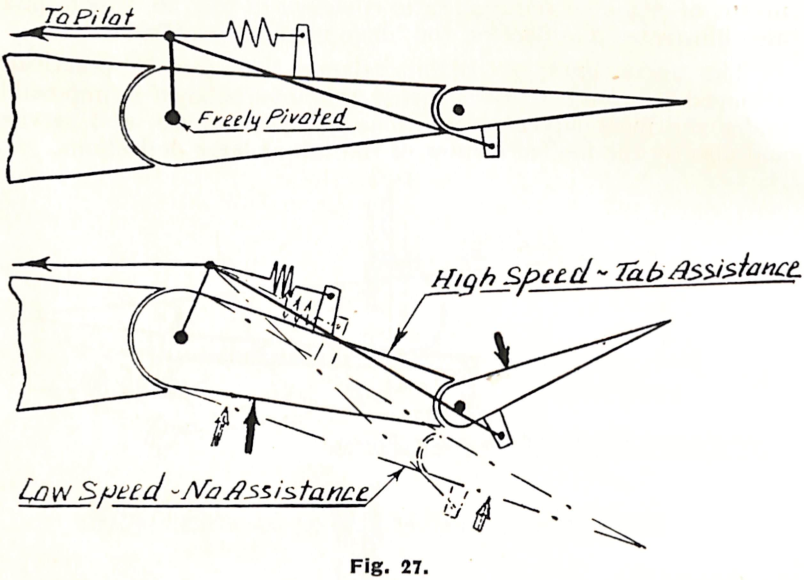


Fig. 27.
The spring tab.

With stick movement the tab deflects in one direction whilst the main control surface moves an amount in the other direction dependent on the elasticity of the spring. Increase of the aerodynamic force on the surface, for instance, with gain in speed, would deflect the spring further with consequent increase of the tab angle relative to the main control, thereby increasing the assistance.

The spring may be preloaded to delay spring tab assistance to higher stick loading, but the modern tendency in design is to

try to avoid introducing such a discontinuity into the hinge moment characteristics.

Mechanical Balance.

Types of mechanical balance include :—

(*a*) Differential Gearing.

(*b*) Variable Gearing.

(*a*) Differential Gearing.

Differential gearing may be used to effect partial 'aerodynamic' balance of ailerons. The better arrangement in this respect is usually when the down going aileron moves through a larger angle than the up going one.

But it has already been shown that the opposite gearing is beneficial from the point of view of adverse yaw.

Thus although some degree of balance may be effected by this means it is usually only at the expense of greater adverse yaw effects. This increase in down going aileron angle may also have deleterious effects on tip stall characteristics.

(*b*) Variable Gearing.

The primary functions and basic principle of operation of a variable geared control mechanism have been discussed in the section on the Control Force and illustrated in Fig. 7.

The geared balance tab, also, is sometimes provided with a variable gear ratio so as to have a greater relative movement at large control deflections.

General Considerations.

Aerodynamic overbalance, or yet even complete balance of a control surface, is to be avoided.

If overbalanced a reversal of stick force occurs, the control surface taking charge and increasing the deflection of its own accord.

If the control surface is totally balanced aerodynamically there is no feel at the pilot's controls.

The need, then, is for the correct degree of partial balance under all flight conditions : an aerodynamic or mechanical balance which provides feel and avoids sloppiness at low speeds and small control surface deflections, a progressive increase in control force with control surface force increment, corresponding with speed and/or deflection increase, to prevent the pilot from inadvertently

imposing undue stresses on the aircraft by manoeuvring it past the structural limits, and yet keeps the maximum control force well within the physical capabilities of the pilot for all functional manoeuvres with no likelihood of complete or overbalance even under the most adverse conditions.

Conditions which might promote overbalance, or, in other words, wander of the resultant aerodynamic force on the control surface forward of the hinge axis, include :—

1. Ice accretion on the control surfaces.
2. Non-linearity of the hinge moment coefficient and control surface deflection curve.
3. Distortion of the control surface in service.
4. Differences in geometry between nominally similar control surfaces from the same production line.
5. Rigging change.
6. Control run stretch.

Preliminary design may provide for the correct partial balance from the outset ; or for complete balance, a degree of anti-balance being incorporated in the control circuit during initial flight tests in the form of easily changeable extended trailing edges or anti-balance tabs.

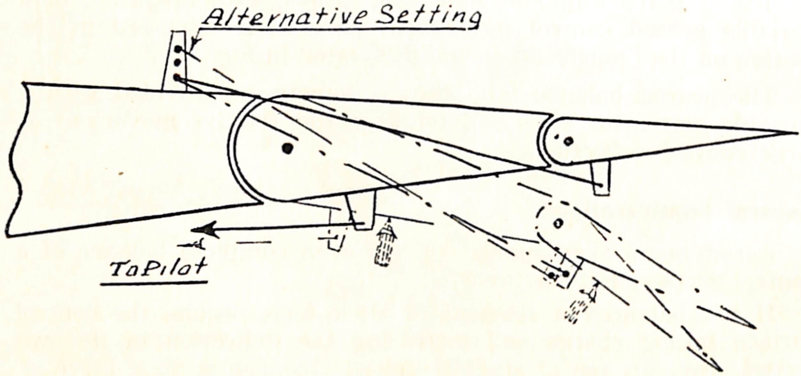


Fig. 28.
The anti-balance tab.

Anti-balance.

A degree of anti-balance may be introduced into a control circuit which is inherently 'light' on the controls and to act as a safeguard against possible over-balance under certain circumstances in flight.

Anti-balance, which increases the effort required of the pilot to deflect the control surfaces, is usually effected by means of an anti-balance tab.

The anti-balance tab, an auxiliary flap inset in the trailing edge of a main control surface, is similar in form to the geared balance tab but of *reverse* gearing. The tab is so coupled that it moves in the same direction as the main control surface. The increment in aerodynamic force introduced by tab deflection produces a hinge moment opposing control surface motion and thus increases the effort required of the pilot for control manipulation. The effect of change in gear ratio is illustrated by the alternative link setting in the anti-balance tab sketch (Fig. 28).

B. **Mass Balance—Flutter.**

The primary reason for mass balance is to prevent, or rather, to reduce the possibility of flutter occurrence. Flutter is an oscillation which may be caused by inertial, elastic or aerodynamic coupling effects. The provision of complete mass balance eliminates the possibility of inertial coupling and hence removes one of the main causes tending to promote control surface flutter.

There are two degrees of mass balance :—

1. Static mass balance.
2. Dynamic mass balance.

Static Mass Balance.

Static mass balance prevents inertia coupling in torsion. It ensures that the centre of gravity of the control surface lies on the control hinge line.

With downward movement of the control surface the lift increment may cause a twisting of the main surface. If the centre of gravity of the control surface lies behind the hinge axis inertia effects cause it to lag behind the motion of the main surface. Torsional reaction eventually stops rotation of the main surface, but by this time the control surface has gained momentum and overshoots. This order of events is illustrated in an exaggerated manner in Fig. 31.

The sequence then continues in the opposite direction and develops into an oscillatory phenomenon known as torsional control flutter.

Static balance ensures, as far as simple torsion is concerned, that both main and control surfaces act in phase.

It is achieved by adding weights ahead of the hinge line.

It exists when the centre of gravity of the control surface lies on the control hinge line. That is, when :—

$$\Sigma\, mx = 0 \qquad (13)$$

where Σ is the Greek letter sigma denoting 'the sum of'
and m is the mass of a small particle distance 'x' from the hinge centre line.

These symbols are illustrated in Figs. 29 and 30.

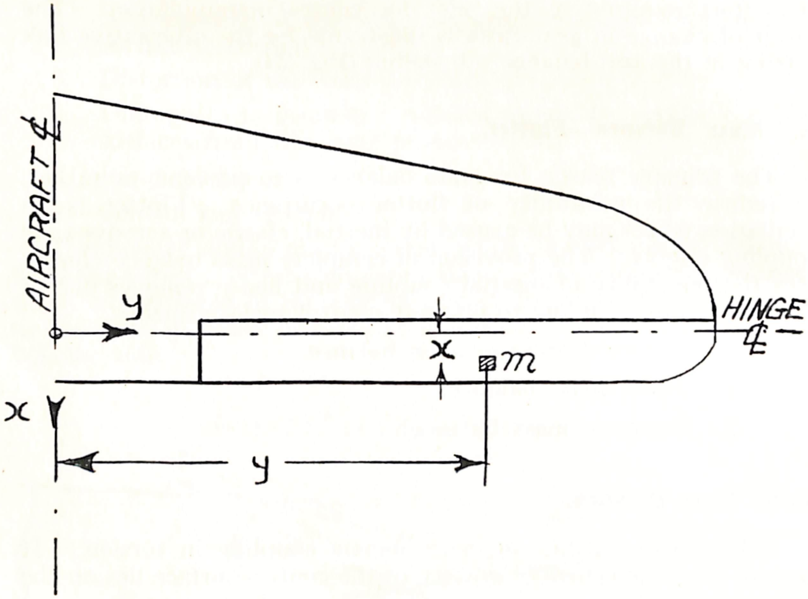


Fig. 29.
Aileron or elevator mass balance notation.

Example 2.

A repair has been done on an aileron. The nett increase in weight due to the patch is $\frac{1}{2}$-lb. at 6″ aft of the hinge line. Find the change in weight needed in the leading edge, 3″ forward of the hinge line to reachieve static mass balance of the aileron.

The criterion is $\Sigma mx = 0$.

Let W lbs. be the mass additive required in the L.E.

Then $+ (\frac{1}{2} \times 6) + (W \times -3) = 0$.

i.e., $W = +1$ lb.

Therefore 1 lb. is to be added at the leading edge.

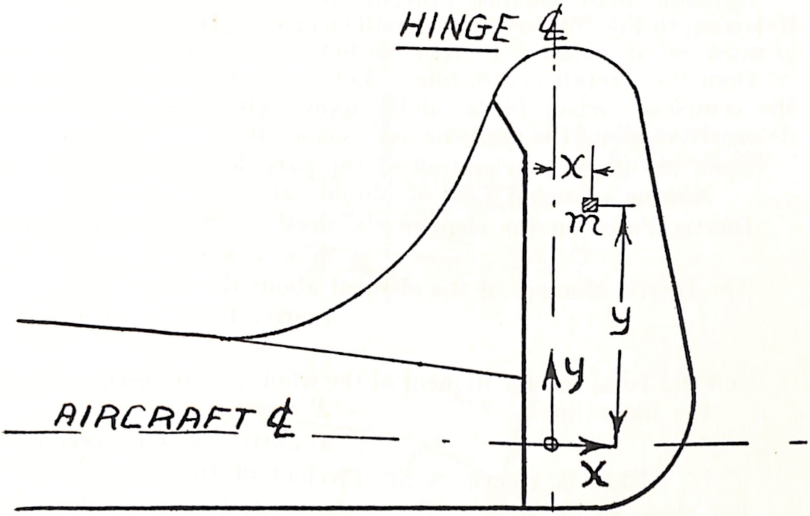


Fig. 30.
Rudder mass balance notation.

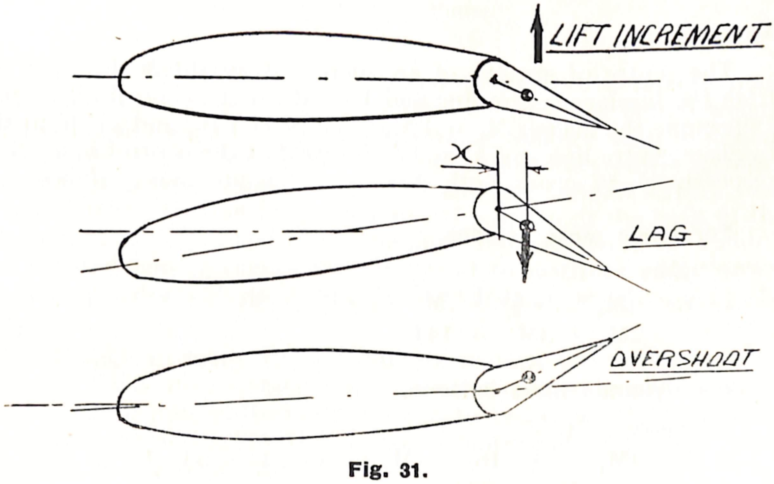


Fig. 31.
Torsional control flutter sequence.

Dynamic Mass Balance.

Dynamic mass balance prevents inertia coupling in flexure. Referring to Fig. 32, consider a small element of the control surface of mass 'm' at distance 'x' from the hinge centre line and distance 'y' from the aircraft centre line. Let the angular acceleration of the complete surface be 'a' and assume that there is no elastic deformation along the span (*i.e.*, 'a' is constant).

Then the linear acceleration of the particle $= a \times y$ and, by Newton's Second Law of Motion, the

Inertia Force on the element $=$ mass $\times$ linear acceleration

$$= m \times a \times y.$$

The Inertia Moment of the element about the hinge line

$$= \text{Inertia Force} \times \text{distance}$$

$$= m \times a \times y \times x$$

and the Total Inertia Moment of the whole control surface about the hinge line $= \Sigma\, maxy$

$$= a\, \Sigma mxy \quad (\text{as } a \text{ is constant}).$$

Σmxy is known as the Product of Inertia.

For there to be no rotation of the control surface when the whole section flexes then the total moment tending to rotate the control must be zero.

$$i.e., \quad \Sigma mxy = 0 \qquad (14)$$

If the product of inertia is zero, or, more practically, slightly negative, the control surface is said to possess dynamic mass balance.

Example 3.

The centre of gravity of an aileron of weight 36 lb. is 20 ft. from the fuselage centre line and 1 ft. aft of the control hinge line. Determine the masses M_1 and M_2 to be placed 16′ and 24′ from the fuselage centre line and $\frac{1}{2}$′ and $\frac{3}{4}$′ forward of the control hinge line, respectively, to ensure both static and dynamic mass balance.

For static mass balance.

$$\Sigma mx = 0.$$

$$i.e., \ (M_1 \times -\tfrac{1}{2}) + (M_2 \times -\tfrac{3}{4}) + (36 \times +1) = 0.$$

$$\therefore \ 2M_1 + 3M_2 = 144 \qquad (A).$$

For dynamic mass balance.

$$\Sigma mxy = 0.$$

$$i.e., \ (M_1 \times -\tfrac{1}{2} \times 16) + (M_2 \times -\tfrac{3}{4} \times 24) + 36 \times 1 \times 20 = 0.$$

$$\therefore \ 4M_1 + 9M_2 = 360 \qquad (B).$$

Solving A and B

$$M_1 = 36 \text{ lbs.} : M_2 = 24 \text{ lbs.}$$

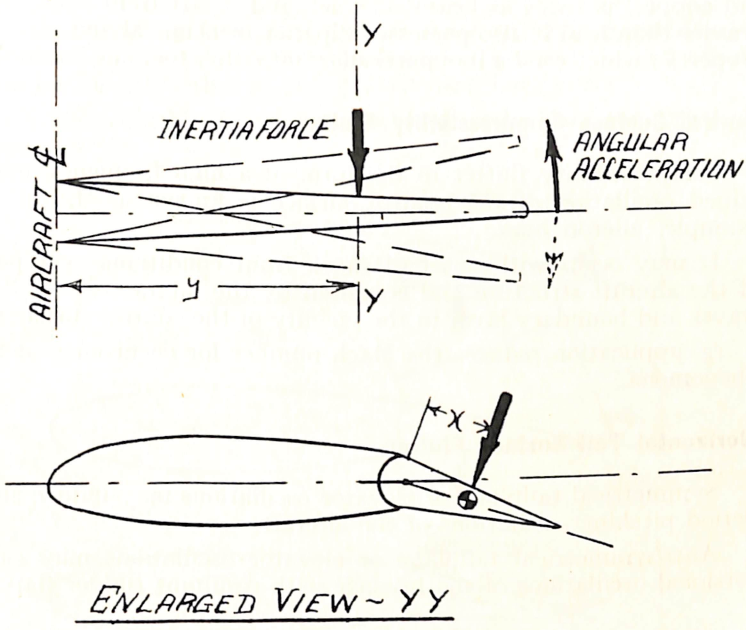


Fig. 32.

Flexural control flutter.

The size of weights to be added to achieve complete mass balance is seen to be large in comparison with the actual weight of the control surface. To achieve mass balance with minimum weight additive the weights should be placed at the greatest distance possible from the control hinge line and the aircraft centre line. That is at the tip and foremost forward position of the span of the elevator and aileron and at the forward top position of the rudder. As isolated masses themselves may tend to institute vibrations it is generally desirable to distribute mass balance weights along the span.

Although in many cases trimming motors and accessories are fitted in the control surface nosing to assist with static and dynamic balancing the bulk of the balancing is achieved by special weighting, weighting according to the design requirement of 'heavy mass addition in a limited space.'

This demands, in contrast to the more usual aeronautical requirement of exceptionally light alloys, the use of high density materials.

The G.E.C. Heavy Alloy (90% tungsten alloyed with nickel and copper) is twice as heavy as steel, and, apart from being 50% heavier than lead it also possesses superior mechanical and physical properties which render it of particular suitability for mass balancing.

Control Surface Compressibility Flutter.

Compressibility flutter in the form of a high frequency maintained oscillation of the control surface is known as 'Buzz', for example, aileron buzz.

It may occur with onset of shock front conditions over parts of the aircraft structure and is caused by the interaction of shock waves and boundary layer in the vicinity of the control surfaces.

'*g*' application reduces the Mach number for occurrence of this phenomena.

Horizontal Tail Surface Flutter.

Symmetrical tailplane or elevator oscillations may induce short period pitching oscillations of the aircraft.

Anti-symmetrical tailplane or elevator oscillations may cause torsional oscillations of the fuselage with resultant rudder flap.

Snaking.

Snaking is an undamped short period yawing oscillation of the aircraft induced by oscillations of the rudder about its hinge axis.

Adequate mass balancing of the rudder reduces the tendency towards snaking. Anti-snaking devices, too, may take the form of lengths of cord or metal strips of angle section (ridges) attached to each side of the rudder trailing edge, or flat metal strips fixed to the rudder trailing edge at right angles to the chord as in the Gloster Meteor and English Electric Canberra.

Tab Flutter.

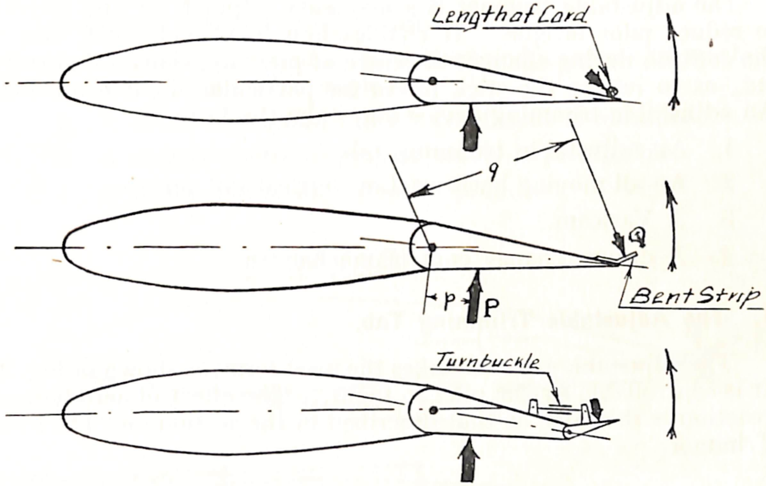
The spring tab is the type most prone to flutter being particularly so disposed when mounted on one side of a single piece elevator. In general, to preclude the possibility of flutter occurrence it is necessary to provide mass balance suitably matched to the spring characteristics; and for the elevator spring tab, in particular, the best arrangement is to have the elevator in separate halves with a spring tab on each section.

Other kinds of tab, if fitted with negligible back-lash in a rigid circuit, may not require mass balance.

X. TRIM.

An aircraft is in a state of trim if for a particular attitude of steady flight that attitude be maintained without control application by the pilot, that is, the aircraft flies 'hands off.'

A trim control, the trimmer, may be of the 'fixed' or adjustable type.



$$Pp = Qq$$

Fig. 33.

Ground adjustment trimming devices.

The Fixed Trimmer (Fig. 33).

A fixed trimmer may take the form of :—

1. A piece of cord attached to the T.E. of the movable control surface.
2. A metal strip (often joggled) attached to the T.E. of the movable control surface.
3. A 'fixed' tab attached to the T.E. of the movable control surface.

Adjustment, which can only be made on the ground, changes the neutral positioning of the control to the desired setting.

The cord trimmer is adjusted by a change of cord length ; the metal strip may be bent to the required bias ; and the positioning of the fixed tab is changed by alteration of the adjustable link length.

With upward bias of the trimmer, as illustrated in Fig. 33, aerodynamic reaction forces the main control surface downwards to the position corresponding with the hinge moment equilibrium, $P_p = Q_q$. The nett effect is additional lift for zero control force increment.

The Adjustable Trimmer.

The adjustable trimmer is a necessary adjunct for any aircraft to reduce pilot fatigue. It enables him to provide such bias to the control, during changes of centre of pressure, centre of gravity, etc., as to relieve the stick forces for particular flight conditions. An adjustable trimming device may take the form of :—

1. An adjustable trimming tab.
2. An all moving horizontal or vertical tail surface.
3. A Varicam.
4. A datum change control mechanism.

1. The Adjustable Trimming Tab.

The adjustable trim tab takes the inset form, as shown in Fig. 34. It is controllable by the pilot in flight. The effect of aerodynamic reaction is the same as that described in the section on "The Fixed Trimmer."

The Trim-cum-Balance Tab (Fig. 34).

The trim-cum-balance tab is one which performs both trimming and aerodynamic balancing functions. For a particular trim setting by the pilot the tab then acts as a normal balance tab, but with a bias.

2. The All-Moving Horizontal and Vertical Tail Surface.

On aircraft incorporating movable tailplanes or fins the trimming system does not often manipulate independent auxiliary surfaces but operates the same surfaces as the primary controls. The power operated action, by an independent actuator, is to provide control surface bias to suit particular flight requirements.

3. The Varicam.

As is suggested by its name the Varicam provides a VARIable CAMber.

It is a movable control surface inset between the fixed section and the rear movable surface as indicated in the sketch of Fig. 34.

Trim actuation, causing variation in incidence and hence camber of the whole section, results in a change of resultant aerodynamic force on the surface which generates a trimming moment about the centre of gravity of the aircraft and thereby alters the neutral setting of the control.

A varicam composes the central portion of the horizontal tail surface of the Lockheed Neptune ; trim actuation taking place *via* an electrically operated irreversible jack.

Advantages claimed of the varicam are :—

1. A smaller horizontal tail surface and a more narrow chord elevator may be employed.
2. A larger maximum tail lift coefficient is obtained.
3. A greater centre of gravity travel is permissible.
4. A lesser 'power-on' effect is produced.

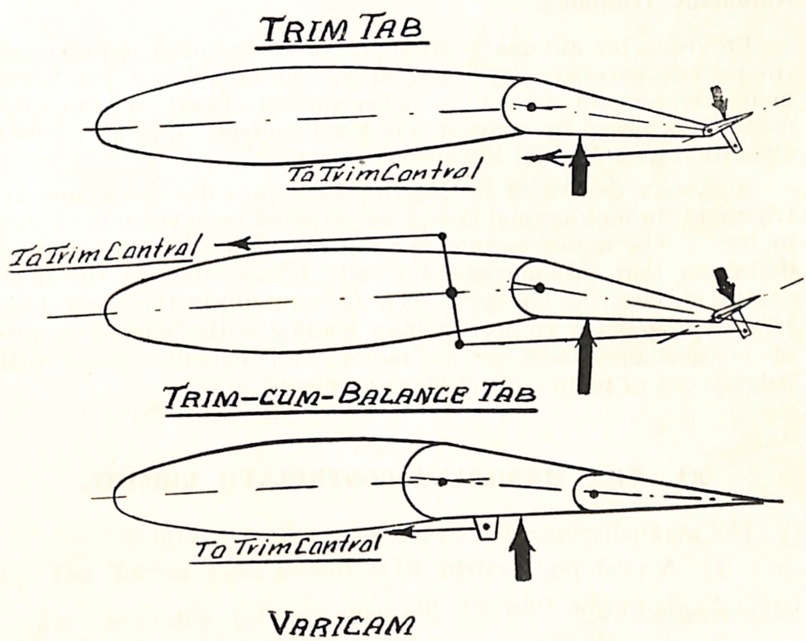


Fig. 34.
Adjustable trimmers.

4. The Datum Change Control Mechanism (Fig. 7).

A power, or hand operated, turnbuckle effects change of datum of the control surface by adjustment of the length of rod (B).

When applied to the operation of elevons datum change is effected by electric motor actuation of worm wheels operating the turnbuckle barrels *via* switches located on the control column : two press button switches providing for lateral trim with a sliding switch governing longitudinal trim.

Trim Actuators.

Trim devices may be operated manually by handwheel or lever, electrically by switches or automatically.

Complete electrical actuation may be incorporated in one switch, the button being slid sideways for lateral trim and fore and aft for longitudinal trim.

Often electrical indicators are used either to show the position of the trim device or to indicate neutral trim.

Automatic Trimming.

Provision for automatic trimming is an essential requirement (to restrict excessive stick forces) for aircraft whose functional manoeuvres might demand a higher rate of change of trim than could be achieved by conventional hand control. (See also Power Operation with Manual Reversion (c)).

A system developed by Boulton-Paul provides for automatic trimming through normal trim tabs actuated by reversible electric motors. The motor current is so governed by the main control deflection that tab movement closely follows that of the main control surface, the lag being such as to retain, in the interests of safety, progressive control column loading with increase in rate of control movement. The motor automatically stops with balance out of main control hinge moments.

XI. THE MANUALLY CONTROLLED CIRCUIT.

The manually operated circuit may take the form of :—

1. A push-pull system.
2. A torque tube circuit.
3. A cable run.
4. A combination of these systems.

1. The Push-Pull System (Fig. 35).

The linkage between the pilot and control surface, or servo tab, consists of compression and tension members, termed push-pull rods. These rods are usually made of similar material to

that of the aircraft structure, namely, aluminium alloy, to eliminate the differential expansions and contractions associated with steel runs in light alloy structures. Circuit direction change is accomplished by the use of bellcranks and lever assemblies.

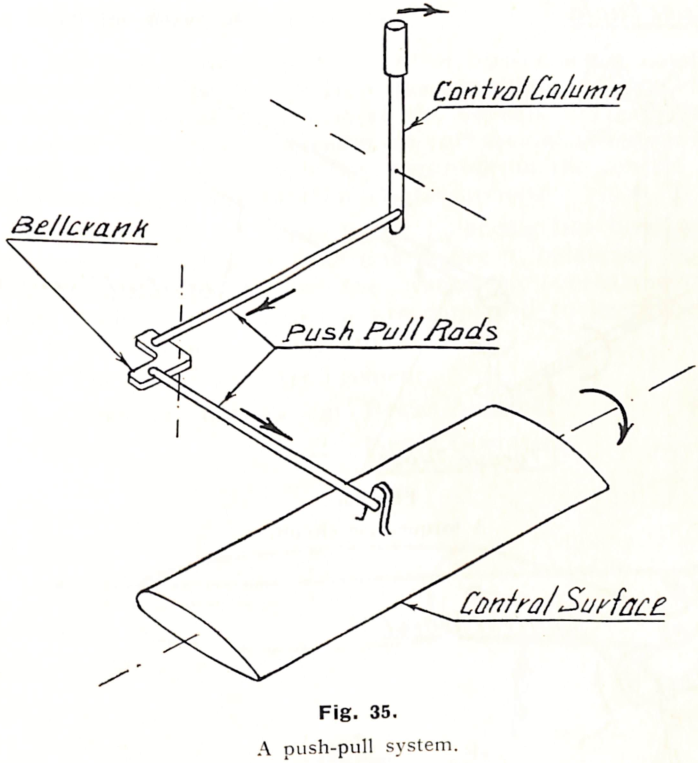


Fig. 35.

A push-pull system.

2. The Torque Tube Circuit (Fig. 36).

In this system torque tubes transmit the drive between pilot and control surface.

3. The Cable Run (Fig. 37).

The control run consists of a continuous circuit of flexible steel cables which operate over pulleys and through fairleads. Differential expansion and contraction is compensated by automatic or manually operated tensioning devices incorporated in the circuit.

4. System Combinations.

Combinations of these systems together with chain and sprocket arrangements may be incorporated in one aircraft, indeed, in one particular control run.

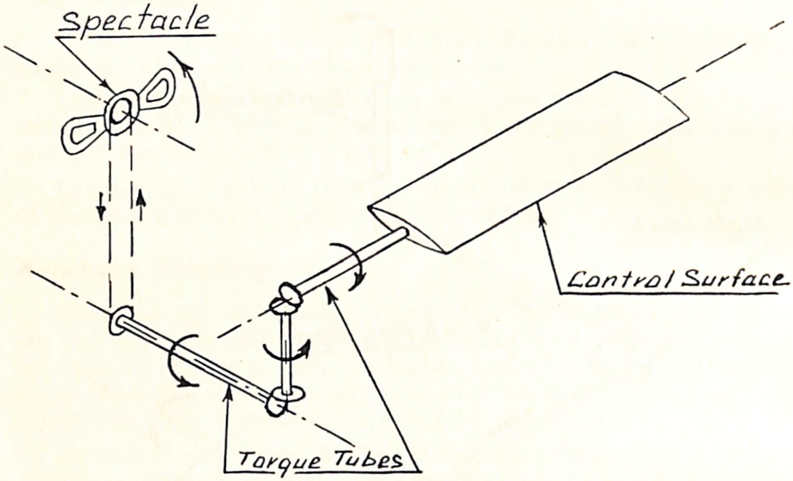


Fig. 36.
A torque-tube circuit.

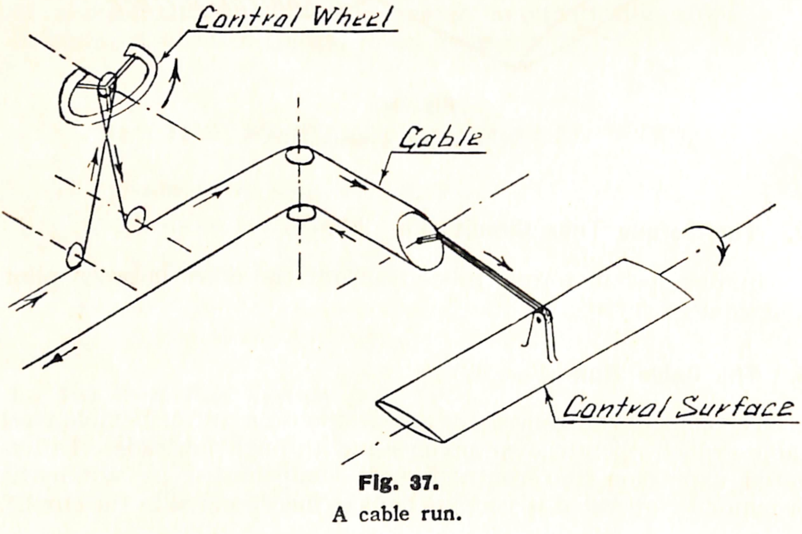


Fig. 37.
A cable run.

XII. SERVO CONTROL.

A servo control is defined as an additional mechanism used to reinforce, or replace, the pilot's effort in working the controls of an aircraft.

The Need for Servo Arrangements.

It has been shown that the control surface hinge moment is proportional both to the square of the equivalent air speed and to the cube of a linear dimension of the aircraft. Therefore with the advent of larger and/or faster aircraft special attention has to be paid to the need to keep the magnitude of the control forces within the range of the pilot's muscular strength.

When the limit for aerodynamic balancing has been reached and there is a limit set by the fine degree of balancing required, with consequent liability of the system to overbalance under certain conditions, other means are employed to keep the stick forces to a low order. These are :—

1. The Servo Tab Arrangement.
2. Powered Controls—(*a*) Power Assisted.
   (*b*) Power Operated.

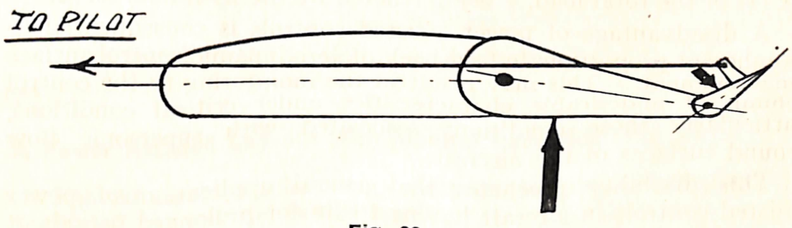


Fig. 38.
The servo tab.

1. The Servo Tab (Fig. 38).

The servo tab is similar in outward appearance to other movable tab arrangements. The pilot, however, has no direct connection with the control surface, the circuit linkage being attached only to the tab.

Tab movement generates an aerodynamic reaction which deflects the main control surface in the opposite direction to such an extent as to restore main control surface hinge moment equilibrium.

Thus movement of the main surface in the desired direction is achieved, by the pilot, by deflection of the tab in the opposite sense. Control forces may be greatly reduced by this means.

The limitation of the servo tab is particularly felt, as is characteristic of all trailing edge controls (ref. Fig. 18), on existence of shock front conditions over surfaces of the aircraft. Under adverse conditions it is possible that insufficient aerodynamic hinge moment would be generated by the servo tab movement to deflect the main control surface, that is, to effect control.

Its chief asset is simplicity.

As with aerodynamic balance arrangements the source of power for servo tab operation is the energy of the airflow.

2. Powered Controls.

(*a*) Power Assisted Controls.

With *power assisted* flying control systems the *pilot's effort* in moving the control surfaces is *reinforced* by power boost.

The proportion of the load taken by the pilot depends on the degree of feed-back, progressive control column feel being retained with increase in aircraft speed and/or control surface deflection.

If the feed-back ratio is 1 : 5 as in the aileron system of the Avro Canada Jetliner, that is, the ailerons are power boosted (hydraulically) in the ratio of 5 : 1 ; the pilot taking one part in six ($\frac{1}{6}$) of the total load, $\frac{5}{6}$ being reacted by the hydraulic servo.

A disadvantage of power assisted controls is consequential to its inherent proportionate feed-back of aerodynamic control surface hinge moment. This may result in the monitoring to the control column of undesirable characteristics under critical conditions, particularly those conditions associated with supersonic flow around surfaces of the aircraft.

This disability precludes the general application of power assisted controls in aircraft having to fly for prolonged periods at transonic or supersonic speeds.

For power assisted controls employing manual reversion as the stand-by, in the event of power failure, certain system requirements are mandatory. These include :—

1. The retention of control surface mass balance to prevent flutter.
2. The retention of robust mechanical control runs of conventional strength and stiffness.
3. The ensurance that the maximum control column loads likely to be encountered in making a safe return and landing after power failure will be within the physical capacity of the pilot.

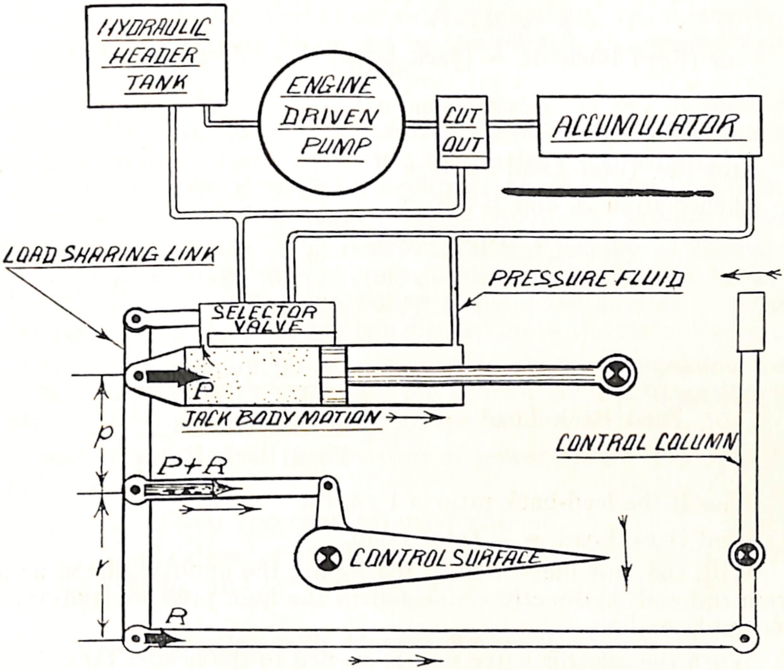


Fig. 39.

A schematic power assisted control layout.

A Power Assisted Lay-out with Manual Reversion (Fig. 39).

Fig. 39 shows a schematic power assisted layout employing a Lockheed 'Servodyne' power unit.

A normal aircraft hydraulic system provides the power.

An engine driven pump supplies, *via* a pre-set automatic cut-out, the hydraulic pressure potential stored in the accumulator.

The control column is connected mechanically to the load sharing and follow up link which also has attachment points to the control surface, movable jack body, and servo control valve.

Movement of the control column changes the setting of the servo selector valve causing the jack body to follow up and assist in the movement of the control surface. When the jack body follow up is complete reclosing of the valve arrests the movement with the control surface assuming an attitude appropriate to the new position of the control column.

The figure shows that :—

$$P \times p = R \times r$$

$$\text{or (Feed Back) } R = \text{(Jack Load) } P \times \text{(Feed Back Ratio)} \; \frac{p}{r} \qquad \text{A.}$$

$$\text{Also the Total Load} = P + R \qquad \text{B.}$$

Thence from A and B :—

$$R = \frac{\text{Total Load}}{1 + \dfrac{r}{p}}$$

$$\text{or, Feed Back Load} = \frac{\text{Total Load}}{1 + \dfrac{1}{\text{Feed Back Ratio}}} \qquad (15)$$

Thus if the feed-back ratio is 1 : 4, the

Feed Back Load $= \frac{1}{5}$ Total Load.

With the unit illustrated in the figure, the annular piston area (ram rod end) is directly connected to the high pressure hydraulic accumulator line.

When the selector valve port is opened to the header tank, jack body motion takes place as shown.

For reverse control column movement the port opens to the high pressure line. The simultaneous application of the same pressure on different areas results in a force differential between the larger piston area and the annular area causing movement of the jack body to the left.

With loss of pressure manual operation of the control surface is facilitated by the almost complete absence of jack stroking loads.

(*b*) **Power Operated Controls.**

With *power operated* flying control systems the *pilot's effort* in moving the control surfaces is *replaced* by servo means.

The pilot has become accustomed to progressive control column feel with increase of speed and/or control surface movement. He desires this state of affairs to continue. It is thus customary to incorporate a synthetic resistance in the circuit to provide artificial control feel.

This synthetic feel unit may be a simple spring to produce control loads proportional to the control surface deflection ; or it

may incorporate pressure head devices to monitor the feel with airspeed V, V^2, etc., according to the functional requirements of the aircraft. It may be pre-set by the pilot to the desired value, and zeroed for the trim condition.

With simple spring feel the incorporation of 'g', or load, restrictors is usual to give progressive control forces with increase of accelerations, thus preventing inadvertent overloading of the aircraft structure through the controls by the pilot. A safety valve may be placed in the power unit circuit for the same reason.

The embodiment of 'g' restrictors is not usually so critical in civil as in military aircraft due to the greater margin between functional manoeuvring and failing loads of the aircraft structure, the restraint of the pilot not being relied on to the same degree.

The safe functioning of a miniature electrical signalling rate control button, with zero feel, for control surface operation *via* power packs depends also upon adequate pilot restraint.

Safety precautions in the event of power failure may take the form of :—

A. A manual reversion arrangement, or

B. Duplication or multiplicity of less reliable parts of the system, and control surfaces.

A. Power Operation with Manual Reversion.

If manual reversion is provided it is necessary to :—

(*a*) Retain mass balance of the control surfaces.

(*b*) Retain a robust mechanical control run.

(*c*) Provide for automatic continuous trim (*e.g.*, electrically operated) to ensure that the aircraft is in a trimmed condition in the event of automatic emergency switch over from power to manual operation. This proviso ensures a smooth change over without the risks associated with sudden unsuspected loading of the control column.

(*d*) Provide automatic cut-out, with power system failure, of artificial feel and power unit stroking loads, to reduce the effort required of the pilot to move the control column.

(*e*) Ensure that the control column loads after failure will not overtax the pilot's strength.

A Power Operated Lay-out with Manual Reversion (Fig. 40).

Fig. 40 illustrates, in diagrammatic form, a power operated Fairey Single Jack Hydrobooster Unit employing manual reversion.

The normal aircraft hydraulic system may provide the pressurized fluid supply to the unit.

Movement of the control column operates the servo selector valve, the pressure directed into the cylinder causing the jack body to follow up and operate the control surface. When the double acting type jack follow-up is complete reclosing of the ports stops the motion with the control surface assuming an attitude appropriate to the new position of the control column.

An overload preventing device in the form of a pressure relief valve, embodied in the servo selector valve, prevents overstressing of the aircraft structure, through the controls, by the pilot.

Power failure automatically withdraws the hydraulically loaded locking pawl and frees the jack ram within the release unit to allow direct manual operation of the control surface, through the jack body, without the need of overcoming jack stroking loads.

A power assisted layout may be obtained with this unit by allowing proportionate feed-back through a similar load sharing linkage to that illustrated in Fig. 39. Whereas a tandem jack version, with an additional hydraulic power supply, is provided for fully duplicated power operated control installations.

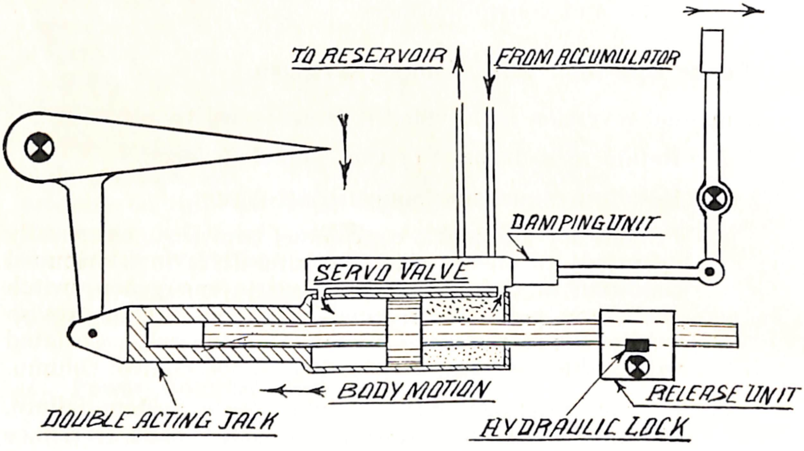


Fig. 40.

A diagrammatic power operated unit with manual reversion.

B. Power Operation with Duplication.

The greatest objection to the use of powered controls is on the ground of reliability. A high degree of reliability may be achieved by incorporating duplicate or multiple power units and power systems, sometimes operating sectional control surfaces.

Advantages of a powered control system with zero feed-back and no manual reversion include :—

(*a*) The dispensing with mass balance of the control surfaces if the system is irreversible and backlash kept to a low order. The consequent weight reduction may more than offset the additional weight of the power packs.

(*b*) The omission of trim tabs, trim being effected by zeroing of the synthetic feel mechanism.

(*c*) Development and testing of the control system, together with pilot acquaintanceship and arbitrary adjustment of the artificial feel mechanism, can be done on the ground prior to the first flight.

(*d*) A saving in weight by using lighter mechanical link-up or electrical connection between some parts of the control run.

(*e*) The possible elimination of the trailing edge flap controls, and the incorporation of all-moving vertical and horizontal tail surfaces and rotating wing tips. These are desirable features of aircraft operating for sustained periods in the transonic and supersonic range.

A Power Operated Lay-out with Duplication (Fig. 41).

Fig. 41 illustrates a power operated lay-out embodying a Boulton Paul electro-hydraulic power pack.

The control column is connected by comparatively light torque tubes and cables, *via* a transmitter unit in which is incorporated an artificial feel and trim device, to the input lever of the servo valve.

A continuously driven electric motor drives the valve operated variable flow hydraulic generator, the speed and direction with which this in turn drives the hydraulic motor depending on the servo valve setting.

The circuit is completed with control surface actuation by a screw jack driven by the hydraulic motor through a differential.

A follow-up shaft returns the servo valve to the neutral position as the control surface assumes the attitude appropriate to the new control column position.

In the event of failure of a unit the hydraulic motor automatically locks, the other unit then operating the control surface, through the differential, at half-speed.

The fundamental difference between powered and non-powered flying control systems is that the energy required for powered operation is stored and transformed internally by fabricated

methods (*e.g.*, engine, motor, electric and hydraulic accumulators, etc.) whereas, for aerodynamic balance and servo tab arrangements the energy source—the dynamic pressure of the relative airflow—is natural and the transformation direct.

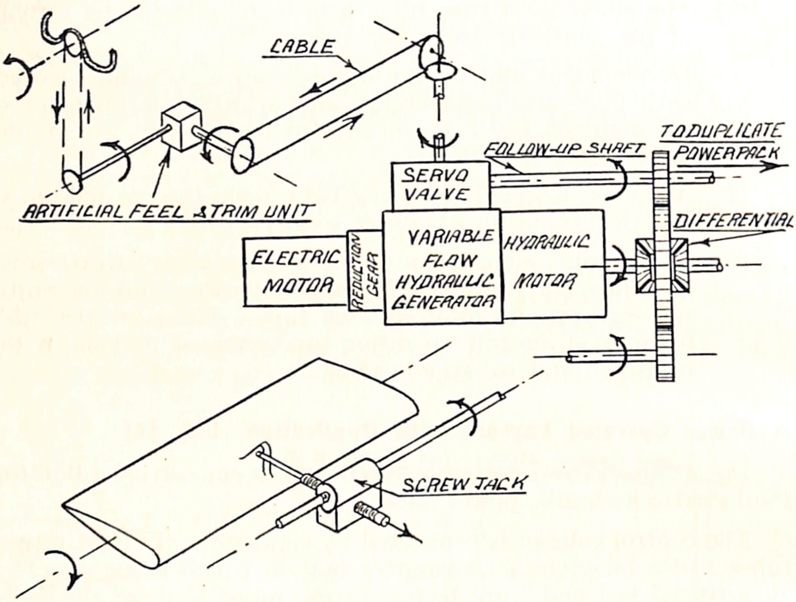


Fig. 41.

A schematic power operated control layout incorporating an electro-hydraulic duplicated power pack.

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23. 1″ ,, ,, ,, ,,
24. $\frac{3}{4}''$ Sq. Steel Tubes as Struts (30 ton yield).
25. $\frac{7}{8}''$,, ,, ,, (30 ton yield).
26. 1″ ,, ,, ,, (30 ton yield).
27. $\frac{3}{4}''$,, ,, ,, (40 ton yield).
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